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JUNE 1971

THERMIONIC SPACECRAFT DESIGN STUDY
120 kw NUCLEAR ELECTRIC PROPULSION SYSTEM

FINAL REPORT

VOLUME I – SUMMARY

June 30, 1971

Prepared Under Contract JPL 952381
for
Thermionic Reactor Systems Project

Propulsion Research and Advanced Concepts Section
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Nuclear Systems Programs
Space Systems
Philadelphia, Pennsylvania 1910

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JPL TECHNICAL MANAGER - C. D. SAWYER

THIS WORK WAS PERFORMED FOR THE JET PROPULSION LABORATORY, CALIFORNIA INSTITUTE OF TECHNOLOGY AS SPONSORED BY THE NATIONAL AERONAUTICS AND SPACE ADMINISTRATION UNDER CONTRACT NAS7-100.

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ABSTRACT

Electrically propelled spacecraft designs for a Comet Halley rendezvous mission, using thermionic reactors as the electrical power source were investigated. Four spacecraft designs were prepared. The four spacecraft designs include two external-fuel reactor concepts (heat pipe cooled diode and independently pumped diode) providing 120 kWe at 40 VDC to the thrust subsystem and two internal-fuel reactor concepts (10 VDC and 40 VDC) providing 120 kWe to the thrust subsystem. The impact of integration with the Space Shuttle, the use of U-233 fueled reactors, alternate EM pumps and main radiator systems is assessed for each of the four spacecraft designs.

The three 40 VDC spacecraft designs are nearly the same size (1.14 m diameter by 20 m to 22 m long) with specific weights from 26 to 30 kg/kWe. The 10 VDC spacecraft design is 27 m long, with a specific weight of about 33 kg/kWe. Integration into the Space Shuttle adds 2 kg/kWe to the 40 VDC spacecraft designs, and 6 kg/kWe to the 10 VDC spacecraft. The use of U-233 fueled reactors reduces the specific weight by 5 kg/kWe for a spacecraft design except the 10 VDC internal-fuel concept.

1.0 SUMMARY

The baseline designs as well as each of the alternate are compared for the Independently Pumped Diode (IPD) and the Heat Pipe Cooled Diode (HCD) external fuel and 10-volt and 40-volt internal fuel reactor spacecraft designs.

1.1 BASELINE REACTOR SPACECRAFT DESIGNS

Design sketches of each of the baseline designs is presented in Figure 1-1. The 10-volt flashlight reactor spacecraft design is the longest, while the IPD reactor spacecraft is the shortest.

Table 1-1 compares the significant spacecraft performance characteristics for each of the baseline designs. The 40 VDC flashlight reactor spacecraft has the lowest propulsion system specific weight of 25.6 kg/kWe compared to the highest propulsion system specific weight of 32.2 kg/kWe for the 10 VDC flashlight reactor spacecraft design. The reactor output power varies from 130.7 kWe for the HCD reactor spacecraft to 166.8 kWe for the 10 VDC flashlight reactor spacecraft.

1.2 ALS-LAUNCHED REACTOR SPACECRAFT DESIGNS

Table 1-2 compares the four reactor spacecraft concepts for launch by the ALS, where the payload bay is limited to 18.3 m. The penalty in propulsion system specific weight for an ALS launch ranges from 1.7 kg/kWe for the HCD reactor spacecraft to 6.9 kg/kWe for the 10 VDC flashlight reactor spacecraft. In order to accommodate the ALS launch, spacecraft diameter must be increased, which increases the propulsion system specific weight.

1.3 U-233 FUELED REACTOR SPACECRAFT DESIGNS

A comparison of the four reactor concepts is shown in Table 1-3 for the alternate designs in which U-233 fuel has been utilized in the reactor diodes. The lightest weight system of all the 120 kWe spacecraft designs considered is the U-233 40 VDC flashlight reactor spacecraft where propulsion system specific weight is 20.8 kg/kWe. Corresponding launch weight is 7580 kg.

FIGURE 1-1

120 KWe COMET HALLEY
REACTOR SPACECRAFT
CONCEPT COMPARISON

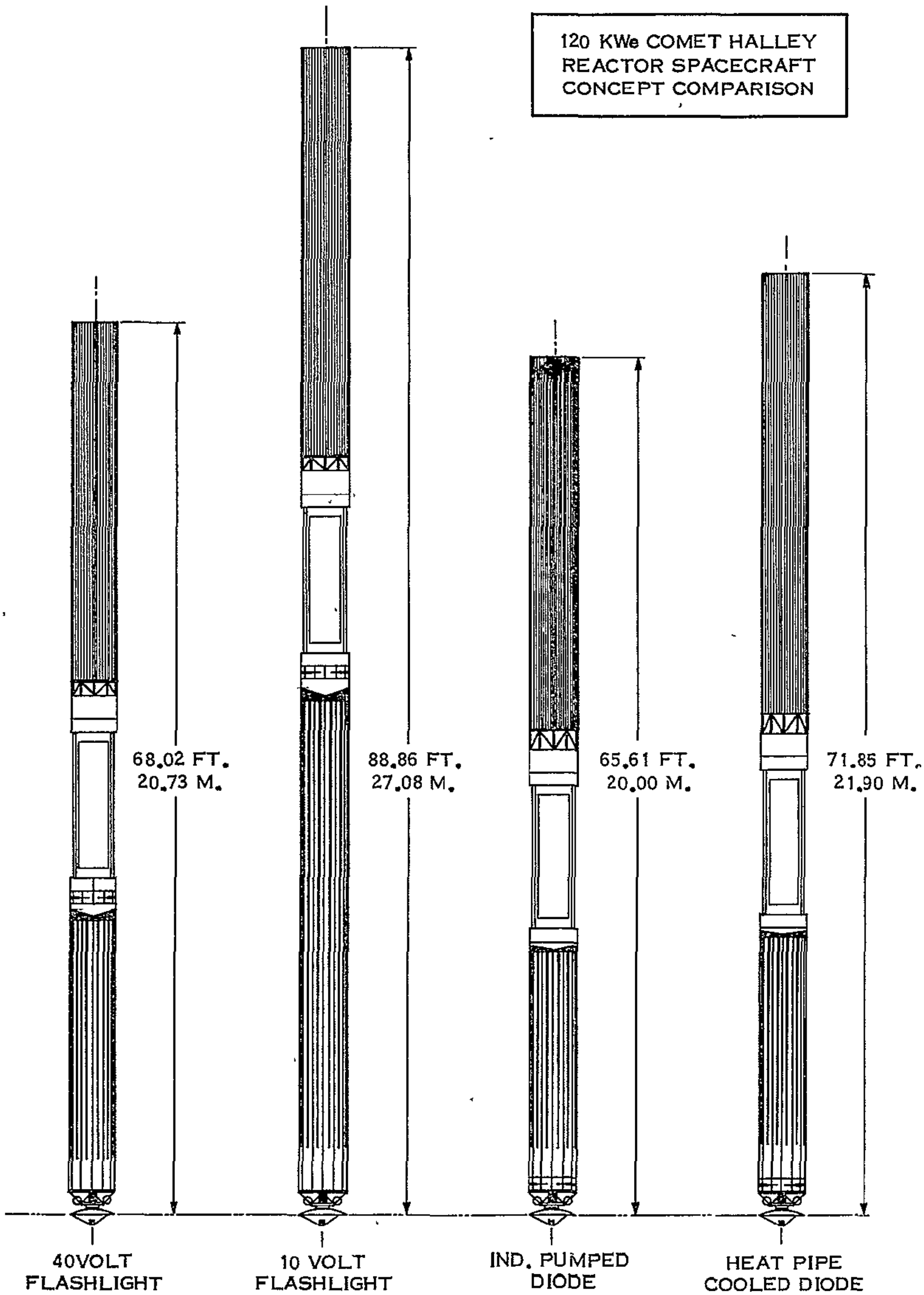


TABLE 1-1

BASELINE SYSTEMS
SPACECRAFT COMPARISON - 120 kWe

				
PARAMETER	EXTERNAL FUEL REACTOR SPACECRAFT		FLASHLIGHT REACTOR SPACECRAFT	
	HEAT PIPE COOLED REACTOR	MULTI-DUCTED EM PUMP COOLED REACTOR	10 VDC SYSTEM	40 VDC SYSTEM
PROPULSION SUBSYSTEM SPECIFIC WEIGHT, kg/kWe	27.7	29.6	32.3	25.6
PROPULSION SUBSYSTEM WEIGHT, kg	3322	3552	3879	3058
LENGTH, m	21.9	21.0	27.1	20.75
DIAMETER, m	1.14	1.14	1.14	1.14
LAUNCH WEIGHT, kg	8411	8690	9280	8188
ION ENGINE INPUT POWER, kWe	109.7	109.7	101.84	109.7
GROSS REACTOR OUTPUT POWER, kWe	130.7	135.7	166.8	136
REACTOR THERMAL POWER, kW	1240	1310	1490	1310
PAYLOAD AT SPACECRAFT END, PERCENT	100	100 (BOOMED 1.0 m)	12	80

TABLE 1-2

ALS LAUNCH CONFIGURATIONS
SPACECRAFT COMPARISON - 120 kWe

				
PARAMETER	EXTERNAL FUEL REACTOR SPACECRAFT		FLASHLIGHT REACTOR SPACECRAFT	
	HEAT PIPE COOLED REACTOR	MULTI-DUCTED EM PUMP COOLED REACTOR	10 VDC SYSTEM	40 VDC SYSTEM
PROPULSION SUBSYSTEM SPECIFIC WEIGHT, kg/kWe	29.4	31.6	39.2	28.0
PROPULSION SUBSYSTEM WEIGHT, kg	3525	3797	4800	3360
LENGTH, m	18.3	18.3	18.3	17.5
DIAMETER, m	1.43	1.31	1.72	1.37
LAUNCH WEIGHT, kg	7957	8228	9114	7790
ION ENGINE INPUT POWER, kWe	109.7	109.7	101.84	109.7
GROSS REACTOR OUTPUT POWER, kWe	130.4	132.7	155	134.7
REACTOR THERMAL POWER, kW	1235	1285	1418	1302
PAYLOAD AT SPACECRAFT END, PERCENT	100 (BOOMED 1.0 m)	100 (BOOMED 2.1 m)	95	100 (BOOMED 1.8 m)

TABLE 1-3

U-233 REACTOR CONFIGURATIONS
SPACECRAFT COMPARISON - 120 kWe

				
PARAMETER	EXTERNAL FUEL REACTOR SPACECRAFT		FLASHLIGHT REACTOR SPACECRAFT	
	HEAT PIPE COOLED REACTOR	MULTI-DUCTED PUMP COOLED REACTOR	10 VDC SYSTEM	40 VDC SYSTEM
PROPULSION SUBSYSTEM SPECIFIC WEIGHT, kg/kWe	23.7	25.5	31.2	20.8
PROPULSION SUBSYSTEM WEIGHT, kg	2842	3067	3760	2488
LENGTH, m	20.9	19.5	27.8	19.1
DIAMETER, m	1.14	1.14	1.14	1.14
LAUNGH WEIGHT, kg	7892	8205	9150	7580
ION ENGINE INPUT POWER, kWe	109.7	109.7	101.84	109.7
GROSS REACTOR OUTPUT POWER, kWe	130.7	135.7	176.6	135.7
REACTOR THERMAL POWER, kW	1240	1310	1505	1307
PAYLOAD AT SPACECRAFT END, PERCENT	70	80	7	27

1.4 DC PUMP REACTOR SPACECRAFT DESIGNS

Table 1-4 compares the performance characteristics of the spacecraft designs which utilized DC pumps in place of the baseline AC pumps. The penalty in propulsion system specific weight by using DC pumps ranges from 0.6 kg/kWe for the HCD reactor spacecraft to 2.3 kg/kWe for the 10 VDC flashlight reactor spacecraft. The baseline IPD reactor spacecraft already uses a multi-ducted DC pump in the baseline design.

1.5 MULTIPLE RADIATOR LOOP REACTOR SPACECRAFT DESIGNS

Table 1-5 compares the alternate spacecraft designs where four independent coolant loops one of which is redundant, replace the baseline single coolant loop. The IPD system is not applicable because the baseline IPD design already has one radiator coolant loop for each diode. For the remaining designs the four-loop radiator was heavier and resulted in longer spacecraft. The least affected spacecraft, the HCD reactor spacecraft, is characterized by a propulsion system specific weight of 28 kg/kWe and spacecraft length of 22.9 m; whereas, the most affected spacecraft, the 10 VDC flashlight reactor spacecraft is characterized by a propulsion system specific weight of 34.4 kg/kWe and spacecraft length of 37.2 m.

1.6 CONCLUSIONS

The major conclusions of this study are:

- The 40 VDC Flashlight and both 40 VDC external fuel spacecraft have the potential to meet Comet Halley Mission performance requirements.
- Within the guidelines and constraints of this study, the U-235 fueled 10 VDC Flashlight spacecraft in the side thrust configuration is unsatisfactory for the Comet Halley mission.
- The U-233 reactor spacecraft designs provide a propulsion system specific weight performance margine of 20 to 30 per cent.

TABLE 1-4

USE OF DC EM PUMPS
SPACECRAFT COMPARISON - 120 kWe

	EXTERNAL FUEL REACTOR SPACECRAFT		FLASHLIGHT REACTOR SPACECRAFT	
				
	HEAT PIPE COOLED REACTOR	MULTI-DUCTED EM PUMP COOLED REACTOR	10 VDC SYSTEM	40 VDC SYSTEM
PROPULSION SUBSYSTEM SPECIFIC WEIGHT, kg/kWe	28.3	29.6	34.6	27.6
PROPULSION SUBSYSTEM WEIGHT, kg	3391	3552	4150	3310
LENGTH, m	23.5	21.0	28.0	23.8
DIAMETER, m	1.14	1.14	1.14	1.14
LAUNCH WEIGHT, kg	8553	8690	9600	8570
ION ENGINE INPUT POWER, kWe	109.7	109.7	103.77	109.7
GROSS REACTOR OUTPUT POWER, kWe	135.3	135.7	176.7	146.6
REACTOR THERMAL POWER, kW	1280	1310	1553	1370
PAYLOAD AT SPACECRAFT END, PERCENT	90	100 (BOOMED 1.0 m)	7	50

TABLE 1-5

ALTERNATE HEAT REJECTION SYSTEMS
FOUR INDEPENDENT RADIATOR LOOPS - ONE REDUNDANT
SPACECRAFT COMPARISON - 120 kWe

		NOT APPLICABLE		
PARAMETER	EXTERNAL FUEL REACTOR SPACECRAFT		FLASHLIGHT REACTOR SPACECRAFT	
	HEAT PIPE COOLED REACTOR	MULTI-DUCTED EM PUMP COOLED REACTOR	10 VDC SYSTEM	40 VDC SYSTEM
PROPULSION SUBSYSTEM SPECIFIC WEIGHT, kg/kWe	28.0		34.4	27.1
PROPULSION SUBSYSTEM WEIGHT, kg	3359		4150	3245
LENGTH, m	22.9		37.2	31.9
DIAMETER, m	1.14		1.14	1.14
LAUNCH WEIGHT, kg	8499		9580	8365
ION ENGINE INPUT POWER, kWe	109.7		101.84	109.7
GROSS REACTOR OUTPUT POWER, kWe	131		167.3	133.5
REACTOR THERMAL POWER, kW	1242		1492	1300
PAYLOAD AT SPACECRAFT END, PERCENT	95		63	100 (EXTENDED 1.8 m)

2.0 INTRODUCTION

A design study investigating thermionic reactor power systems for nuclear electric propelled unmanned spacecraft was initiated by the General Electric Company on February 4, 1969, for the Jet Propulsion Laboratory under Contract No. JPL 952381. The first phase of this effort was directed toward the design definition of a nominal 300 kWe (gross) thermionic reactor spacecraft for a Jupiter Orbiter Mission. The impact of both internal fuel (flashlight) and the external fuel thermionic reactor concepts on the spacecraft configuration, performance and weight was assessed. The results of this phase of the study are reported in Reference 2-1.

The investigation of thermionic spacecraft designs directed for the Comet Halley rendezvous mission, at a nominal power level of 120 kWe for 10,000 to 15,000 effective full power hours, was initiated in October 1970. The reference spacecraft designs are based upon the external fuel thermionic reactor and the internal fuel (flashlight) thermionic reactor designs.

The reference launch mode is direct injection to earth escape with the Titan-Centaur class launch vehicle. In addition, the spacecraft configuration shall meet the probable geometry constraints imposed by the Advanced Logistic System.

The baseline spacecraft arrangement is a high L/D ratio ($\sim$20), side thrust configuration with the thrust array in the center of the spacecraft and the thrust vector perpendicular to the axis of the spacecraft. This arrangement separates the high temperature components (1000°K nominal), the reactor and main radiator, from the low temperature (367°K nominal) power conditioning and payload components. Of equal importance, the science payload and communications subsystems can be located at one end of the spacecraft where there is almost a 3π steradian field of view. A 4π steradian field of view may be achievable by rotating the spacecraft about the thrust axis.

The major objective of this study is to investigate and provide a well defined Nuclear Electric Propulsion (NEP) system spacecraft to be used in NEP mission trajectory analysis, and in defining the

thermionic reactor technology effort. The key performance elements are the propulsion system lifetime and specific weight. The specific weight, α, is defined as the propulsion system mass, M_{ps}, divided by that electric power, P_e, that is delivered to the thrust subsystem.

3.0 OBJECTIVES, GUIDELINES AND COMMON SUBSYSTEMS

This section outlines the study objectives, design guidelines, and subsystems that are common to all spacecraft designs considered in this study.

3.1 OBJECTIVES

The primary study objective is the definition of a 120 kWe (P_e) thermionic unmanned electric propulsion spacecraft design to perform the Comet Halley rendezvous mission. The baseline powerplant designs shall deliver 120 kWe power to the thrust system for 10,000 to 15,000 effective full power hours. The design weight objective for the 120 kWe electric propulsion system is 2725 kg (6000 lbs).

The evaluation of the external fuel reactor will investigate the relative merits of a heat pipe cooled reactor, and a multi-ducted DC EM pump cooled external fueled reactor design.

This study will also assess the impact of the flashlight in-core thermionic reactor, being developed by Gulf General Atomics for the USAEC, upon the Comet Halley spacecraft weight and configuration. The scope of this effort includes an assessment of both 10 VDC and 40 VDC flashlight reactor designs.

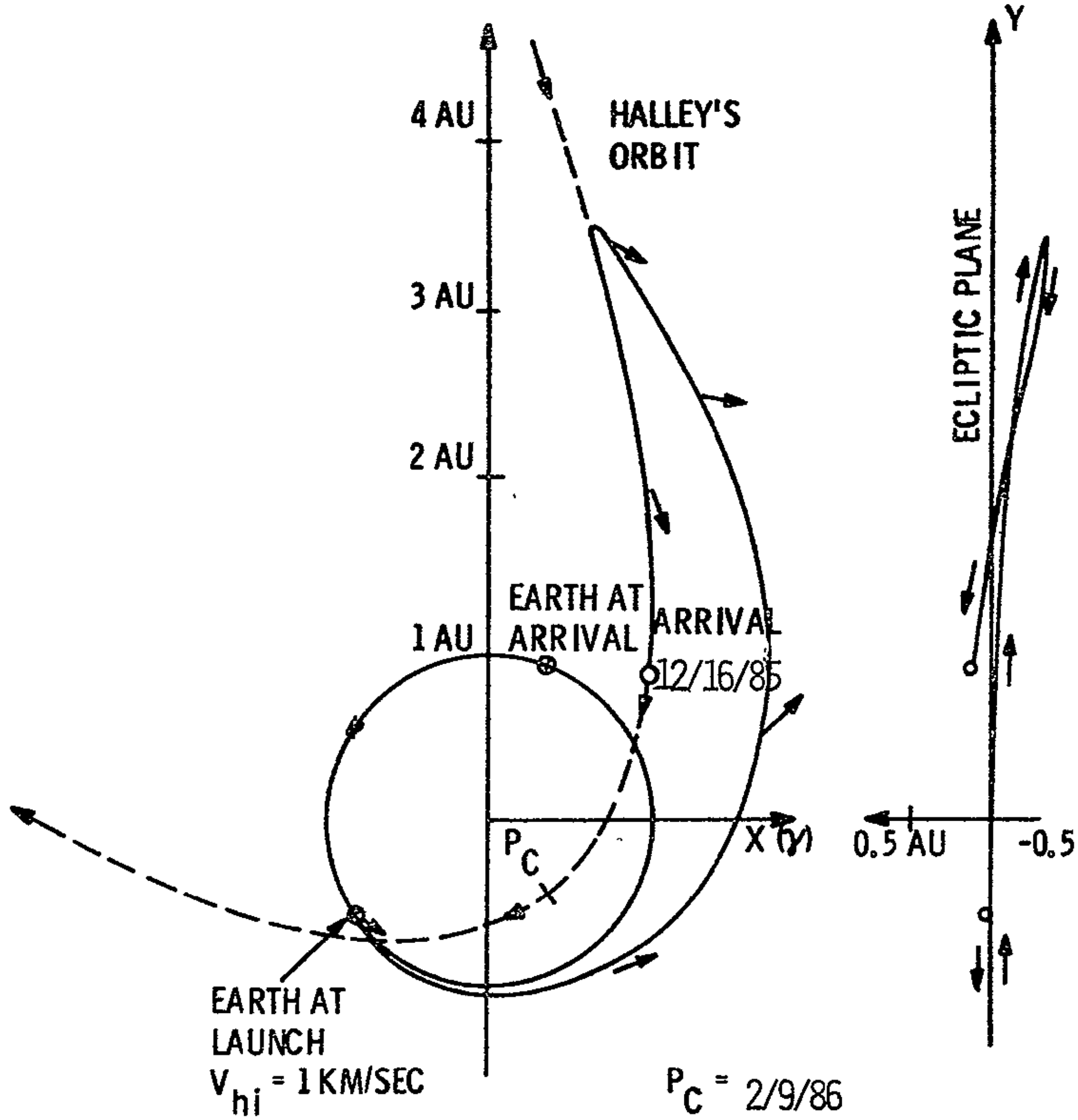
3.2 COMET HALLEY MISSION

The 940-day Comet Halley rendezvous mission has been selected as the baseline mission for the spacecraft definition.

Since rendezvous with Comet Halley is an extremely difficult mission to perform because of its unique retrograde motion about the sun, this mission presents an attractive opportunity for nuclear electric propulsion. The spacecraft trajectory, illustrated in Figure 3-1 (Reference 3-1) is based on arrival of Halley's Comet on December 16, 1985, which is 55 days before perihelion passage. At this point the geocentric distance is near minimum and the comet brightness as seen from Earth is about 4th magnitude. The 1983 launch is accomplished by a Titan/Centaur class launch vehicle, which injects the spacecraft beyond Earth escape with a nearly optimum hyperbolic excess velocity of approximately 1 km/sec. The spacecraft thrust vector rotates slowly clockwise with respect to the sun line. Retrograde motion

FIGURE 3-1

940-DAY RENDEZVOUS TRAJECTORY TO HALLEY'S COMET NUCLEAR-ELECTRIC LOW THRUST FLIGHT MODE



begins near the trajectory aphelion (3.5 AU), which occurs about 600 days after launch. Comet Halley is about 5.2 AU from the sun at this time and is rapidly approaching the spacecraft.

The baseline mission characteristics, which have been selected in conjunction with JPL from the matrix presented in Table 3-1 (Reference 3-2), include a specific impulse of 6000 seconds, net payload of 664 kg, and power level of 120 kWe supplied to thrust subsystem. Launch capability of the Titan/Centaur corresponding to 1 km/sec hyperbolic excess velocity is 7000 kg. This baseline mission provides a useful payload and a challenging full power thrusting time of 16,000 hours. The mercury propellant is utilized in the spacecraft design to meet the 10^7 rad integrated gamma dose limit without requiring permanent gamma shielding, with its associated weight penalty.

The mission analysis (Reference 3-1) employed an electric propulsion system specific mass, α, of 28.3 kg/kWe at the 120 kWe P_e level, where α is the electric propulsion system mass divided by the electric power delivered to the thrust subsystem, P_e.

The electric propulsion system mass, Mps, includes the mass of the reactor, radiation shield, radiators, power conditioning, pumps, electric thruster array and integrating structure.

3.3 DESIGN GUIDELINES

The following design guidelines have been established in conjunction with JPL as a reference for the design definition of thermionic electric propulsion spacecraft:

(a) The baseline mission is a 940-day Comet Halley rendezvous. After injection to Earth escape by the Titan/Centaur class launch vehicle, the following mission times (Reference 3-2) and power levels have been established:

Mission Mode	Power Level Gross, kWe	Time, Days
Initial Thrust	140	174
Coast	20	274
Final Thrust	140	492
Rendezvous	20	155

TABLE 3-1

NUCLEAR ELECTRIC HALLEY RENDEZVOUS MISSION PARAMETERS, 940-DAY FLIGHT TIME

		SPECIFIC IMPULSE, SECONDS				
Pe kWe		5000	6000	7000	8000	9000
V_b VOLTS		3100	4000	5500	7100	9000
100	M_n	495	780	⊠	⊠	⊠
	M_p	3780	3550	⊠	⊠	⊠
	T_p	658	841	⊠	⊠	⊠
120	M_n	315	$\boxed{664}$	1340	⊠	⊠
	M_p	3660	$\boxed{3330}$	3180	⊠	⊠
	T_p	532	$\boxed{666}$	827	⊠	⊠
140	M_n	89	470	730	908	⊠
	M_p	3600	3230	2980	2820	⊠
	T_p	447	553	677	819	⊠

P_e ELECTRICAL POWER INPUT TO THRUST SUBSYSTEM, kWe

M_n NET SPACECRAFT MASS DELIVERED, EXCLUDING ANY ULLAGE OR UNCERTAINTY ALLOWANCE, KG

M_p REQUIRED PROPELLANT MASS, EXCLUDING ANY ULLAGE OR UNCERTAINTY ALLOWANCE, KG

T_p PROPULSION TIME, DAYS

FLIGHT TIME: 940 DAYS

LAUNCH VEHICLE: TITAN III F/CENTAUR, 16700 LBS INJECTED MASS PAYLOAD CAPABILITY AT A HYPERBOLIC LAUNCH VELOCITY OF 1.0 KM/SEC

⊠ MISSION IMPOSSIBLE: INSUFFICIENT ACCELERATION FOR FLIGHT TIME

(b) Net spacecraft weight is 664 kg.

(c) Thrust is provided by 30 mercury ion thrusters (including 6 spares) weighing 213 kg, including the thrust vector control system.

(d) Equivalent full-power reactor operating time is 17,500 hours.

(e) Net power of 120 kWe shall be supplied to the thrust subsystem.

(f) The mercury propellant requirements to accomplish the mission is 3660 kg, including 10 percent allowance for ullage and other uncertainties.

(g) Beryllium fin or beryllium fin, stainless steel tube radiators will be employed.

(h) Radiator non-puncture probability is 0.95 for the 940-day mission. Armor requirements are based on a deep space meteoroid flux that is 43 percent that of Earth orbit. The meteoroid protection requirement will be compatible with the following models:

I. Penetration Model

$$t = 0.5\, m^{0.352} \rho_m^{1/6} v^{0.875}$$

where

t = armor thickness, cm
ρ_m = meteoroid density, gm/cm^3
m = meteoroid mass, gm
v = meteoroid velocity, km/sec

II. Meteoroid Flux

$$\phi = \alpha m^{-\beta}$$

where

ϕ = cumulative meteoroid flux, number particles/m^{2}sec

α = empirical coefficient

β = empirical exponent

m = meteoroid mass, gm

III. Probability of Penetration

The non-puncture probability is,

$$P_{(0)} = e^{-\phi AT}$$

where

$P_{(0)}$ = non-puncture probability

ϕ = cumulative meteoroid flux, number particles/m^{2}sec

A = projected vulnerable area of the spacecraft (radiator), m^2

T = exposure time, seconds

The baseline data listed below is used in conjunction with the previous models to calculate an equivalent near Earth meteoroid protection requirement:

$$\bar{\rho}_m = 0.5 \text{ g/cm}^3$$

$$\bar{v} = 20 \text{ km/sec}$$

$$\alpha = 6.62\ (10)^{-15}$$

$$\beta = 1.34$$

$$P_{(0)} = 0.95$$

$$T = 7.2\ (10)^7 \text{ sec (20,000 hr)}$$

Then, an effective thickness, t_{eff}, for the mission to Comet Halley may be calculated from

$$t_{eff} = 0.432t$$

The radiator models used in this study have been developed from the SPARTAN II computer code (Reference 3-3) results and are based on the preceding near Earth meteoroid protection requirement.

(i) Maximum allowable electronic component temperature is 367°K.

(j) Mean sink temperature for the entire mission is 166°K.

(k) Each diode in the external fuel reactor is cooled by an independent loop in parallel with all other diodes, (hydraulic), for the multiducted EM pump cooled design, or by a single loop heat exchanger for the heat pipe cooled reactor concept.

(l) Maximum allowable neutron shield temperature is 812°K.

(m) Maximum allowable EM pump winding temperature is 644°K.

(n) Reactor controls power requirement is 0.2 kWe.

(o) Payload power allowance is 1.0 kWe.

(p) Cesium reservoir temperature control power requirement is 0.5 kWe.

(q) All liquid metal coolants are NaK-78.

(r) Science payload, power conditioning, and communications will be shielded to within an integrated dose of 10^{12} nvt ($\geq$ 1 Mev) and 10^7 rads gamma. Credit will be taken for attenuation by non-shielding materials.

(s) No permanent gamma shielding will be employed in the baseline spacecraft designs.

(t) Although it may be practical to fold the spacecraft to facilitate shuttle integration, the option will not be employed in this study.

(u) Since reliability of the individual components is unknown at this time, a mission reliability number was not established. Emphasis is placed on redundancy, launchable configurations, conservative mechanical engineering design, and good engineering judgment.

3.4 LAUNCH VEHICLES

The baseline launch vehicle selected for the thermionic reactor spacecraft design study is the Titan IIID7/Centaur*. An alternative design has been generated for launch of the thermionic reactor spacecraft by the Advanced Logistics System (ALS). This section presents characteristics of the ALS and the Titan III D7/Centaur as well as growth versions of the Titan III/Centaur family.

3.4.1 TITAN III

The characteristics of the Titan III launch vehicles that are being considered for launch of a nuclear electric spacecraft for the Comet Halley rendezvous mission are presented. Data are included for the following launch vehicles (Reference 3-4):

- Titan IIID7/Centaur
- Titan IIIL2/Transtage
- Titan IIIL2/Centaur
- Titan IIIL4/Transtage
- Titan IIIL4/Centaur

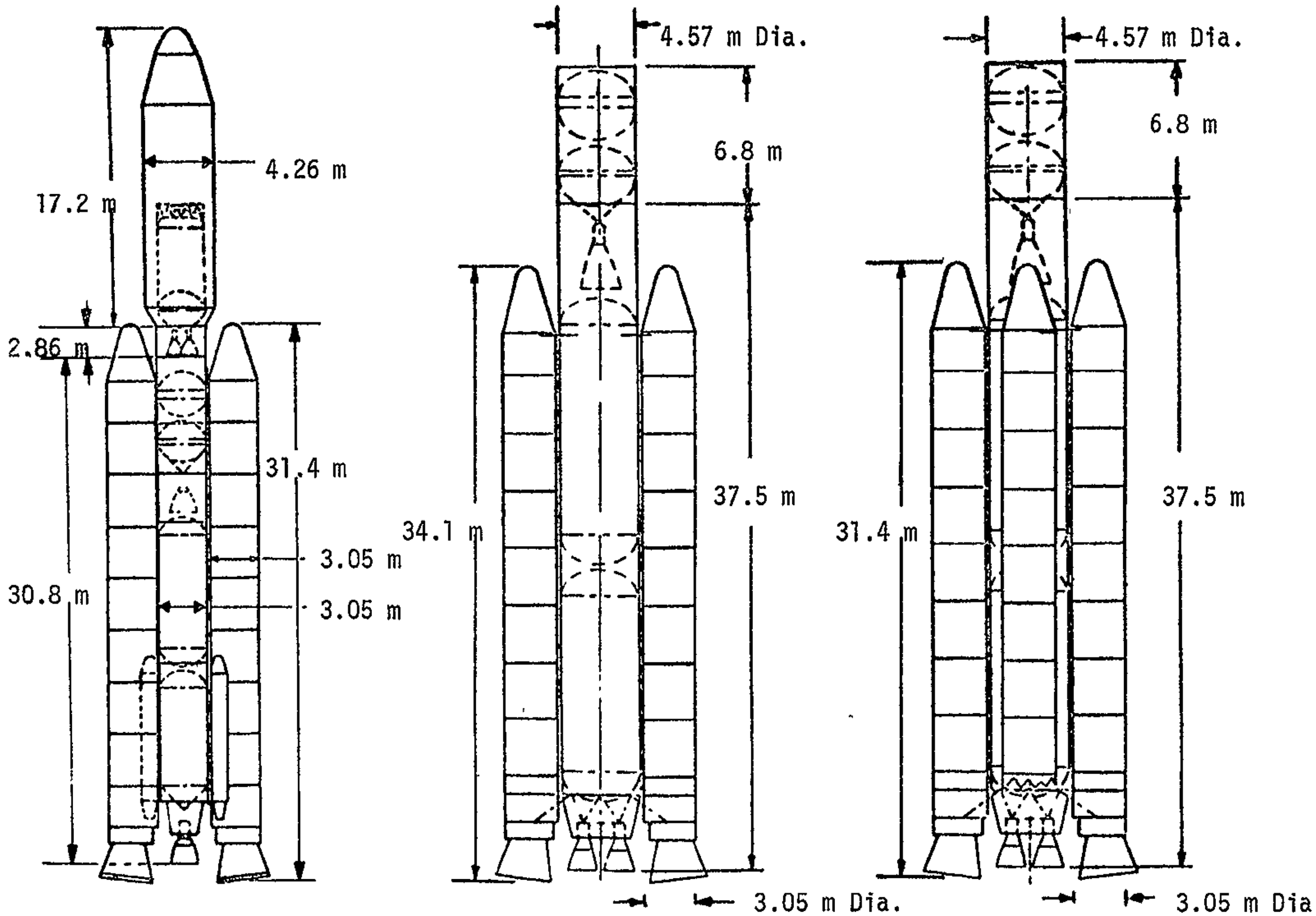
A schematic diagram of each of the core launch vehicles (without Centaur or Transtage upper stage) is shown in Figure 3-2.

Payload performance of the five launch vehicles is presented in Figure 3-3 for the range of characteristic velocity and corresponding hyperbolic excess velocity of interest to the Comet Halley mission. The payload that is indicated on these curves was calculated after assuming the payload fairing weights shown in Table 3-2. Also, payload was maximized by assuming a 90° (directly East) launch azimuth from ETR.

*Synonymous with Titan IIIF/Centaur and Titan IIIC7/Centuar.

FIGURE 3-2

TITAN IIID7, IIIL2, AND IIIL4 LAUNCH VEHICLES



TITAN IIID7/CENTAUR
STRETCHED CORE
TWO 7-SEGMENT SRM

TITAN IIIL2
LARGE DIAMETER CORE
TWO 7-SEGMENT SRM

TITAN IIIL4
LARGE DIAMETER CORE
FOUR 7-SEGMENT SRM

FIGURE 3-3

TITAN III LAUNCH CAPABILITY

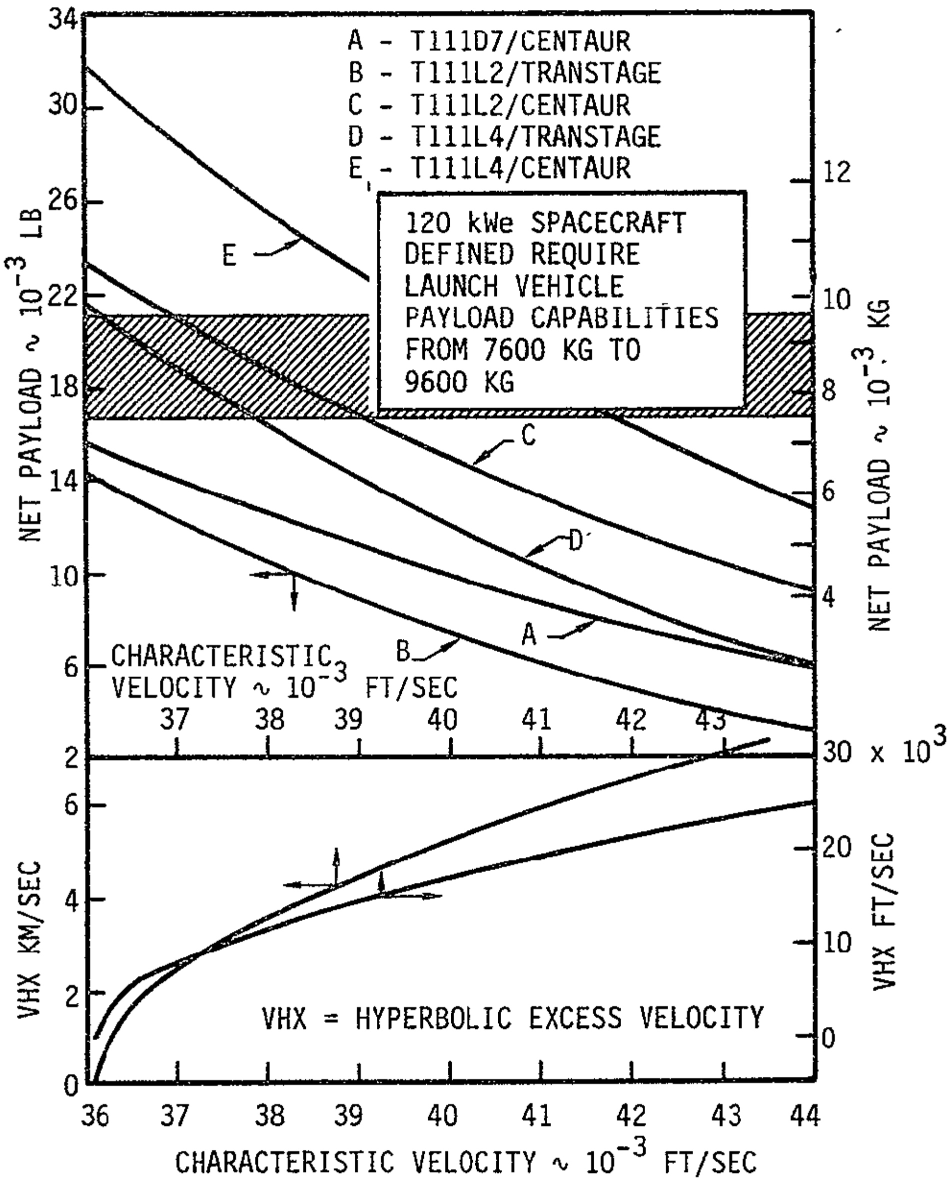


TABLE 3-2

PAYLOAD FAIRING WEIGHTS

PAYLOAD FAIRING			
UPPER STAGE	JETTISON WT. kg	DIAMETER m	TOTAL LENGTH m
Transtage	892.1	3.05	7.62
Centaur	1741.5	3.35	15.54

In addition to payload capability, launch vehicle characteristics that affect integration of the spacecraft with the launch vehicle have been obtained (Reference 3-4). These include:

- Maximum lateral acceleration of 1.5 g's during launch.
- Maximum longitudinal acceleration of 6 to 7 g's in compression and 2.5 g's in tension during launch.
- For additional required payload fairing, payload penalty is 0.08 pounds of payload per pound of fairing.

The shaded band on Figure 3-3 indicates the launch vehicle payload requirements for the 120 kWe side thrust thermionic spacecraft established in this study. These range from approximately 7600 kg to 9600 kg. The 9000 kg capability of the reference Titan IIID7/Centaur launch vehicle cannot meet these requirements. However, Figure 3-3 demonstrates that growth versions of the Titan III/Centaur launch vehicle will meet the range of thermionic spacecraft weights characterized for the Comet Halley mission in this study.

3.4.2 ADVANCED LOGISTICS SYSTEM

For both the external fuel and flashlight reactor spacecraft, the baseline designs have been reconfigured for launch by the Advanced Logistics System (ALS). This subsection identifies the major differences between the TIIID7/Centaur launch vehicle and the ALS.

Mission profile for the launch of a nuclear electric propulsion spacecraft by the ALS is shown schematically in Figure 3-4. Although payload configurations of 18.3 m and 9.2 m in length are presented in Figure 3-4, only the payload length limitation of 18.3 m was assumed for this study.

Current baseline ALS design point characteristics as outlined in Reference 3-5 are presented in Table 3-3 for 90° azimuth (directly East), resupply, and polar launches from ETR. Structural integration of the spacecraft with the launch vehicle is based on maximum steady state load factors of 3 g longitudinal acceleration and 1 g lateral acceleration.

3.5 THRUSTER ARRAY

The thruster array consists of the ion engines, the Thrust Vector Control (TVC) system, and their immediate support structure. The thruster array delineated for the 120 kWe Comet Halley rendezvous spacecraft is based upon the hardware and analytical techniques being developed for solar electric propulsion spacecraft (Reference 3-6). Structure weights for the thruster array are based on design layouts completed for the 120 kWe side thrust nuclear electric propulsion spacecraft. The thruster array presented is common to all four baseline spacecraft designs included in this report.

FIGURE 3-4

120 kWe THERMIONIC SPACECRAFT
SHUTTLE INTEGRATION

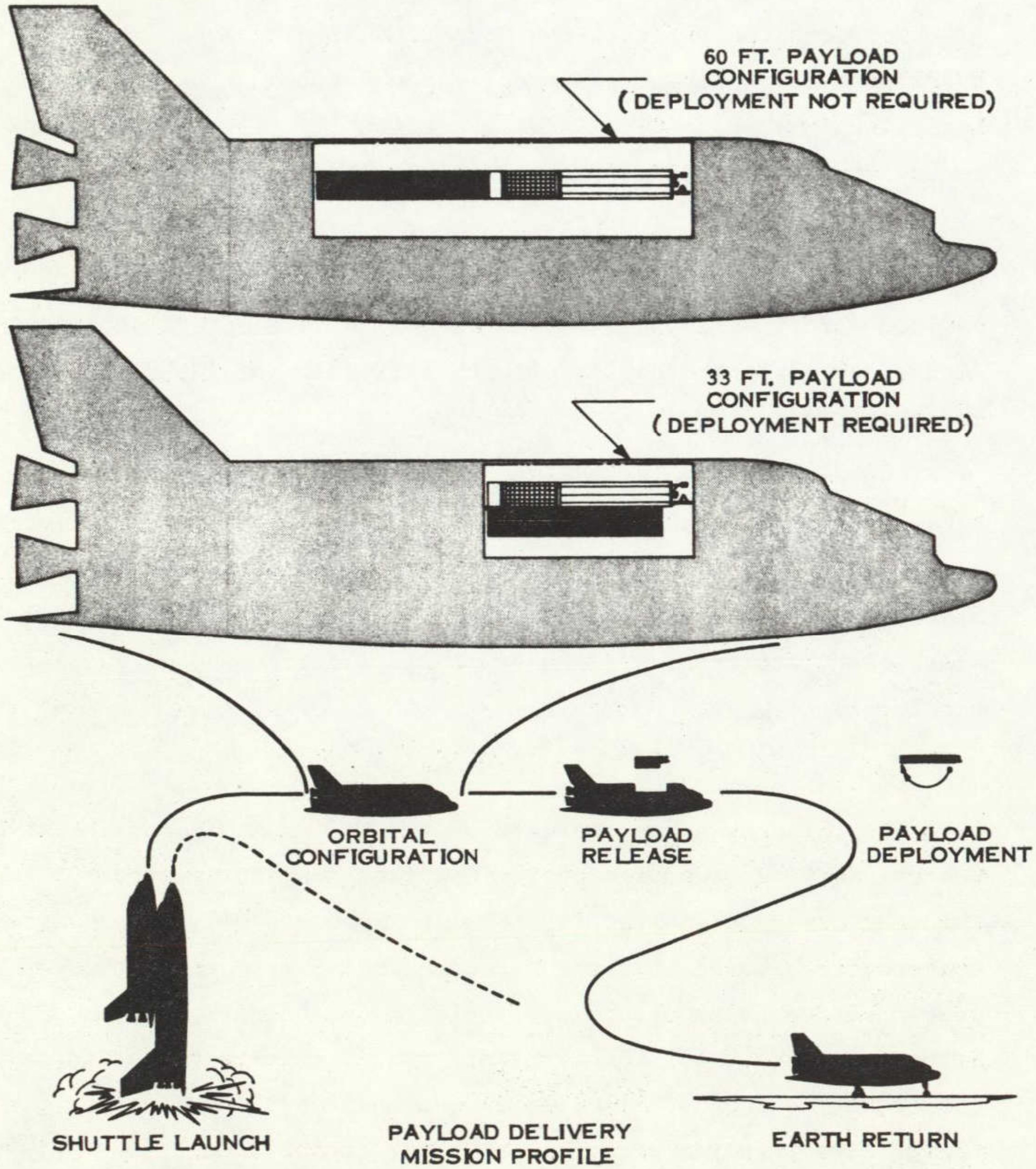


TABLE 3-3

BASELINE ALS DESIGN POINT CHARACTERISTICS

PARAMETER	MISSION		
	Easterly	Resupply	Polar
Payload, kg	36,162	14,107	18,160
Gross Liftoff Weight, kg			
System	2,105,900	2,099,635	2,090,076
Booster	1,715,739	1,716,665	1,718,517
Orbiter	390,160	382,970	371,559
Uncert/Growth Reserve, Per-Cent	9.5	9.5	9.5
Orbiter OMS Capability km/sec	0.206	0.457	0.218
Staging Velocity, km/sec			
IDEAL	4.65	4.68	4.72
REAL	3.31	3.34	3.38

3.5.1 PARTICULAR GUIDELINES AND CONSTRAINTS

The definition of the number, size and arrangement of the electron bombardment mercury ion engines must consider the following guidelines and constraints:

- 120 kWe P_e is delivered to the main power conditioning for distribution to all operating ion engines.
- The number of ion engines must include 20 percent redundancy.
- The minimum dimension of the thruster array is dictated by the spacecraft diameter. In the side thrust spacecraft, this diameter is generally defined by the mercury propellant tank configuration which meets the mission gamma shield requirements.
- The width of the thruster array must permit a translation of the entire array a distance equal to one-half of the beam diameter of the ion engine in the direction perpendicular to the spacecraft axis, without extending the thruster array outside the shadow of the radiation shield.
- The axial length provided for the thruster array must permit a translation of the entire thruster array a distance equal to one ion engine beam diameter in either direction parallel to the axis of the spacecraft. Greater translation along the axis of the spacecraft is provided to assure effective control of the high L/D of the side thrust spacecraft.
- A reasonable number of the ion engines must be gimbaled to provide for roll TVC about the thrust axis. (Pitch and yaw control are achieved by translation of the entire thruster array, noted above.) The ion engine spacing must permit rotation of the gimbaled ion engines $\pm$ 10 degrees. The spacing of gimbaled ion engines requires special consideration in order to accommodate the gimbal mechanism, based on designs being developed for Solar Electric Propulsion (SEP).
- The number and size of the ion engines must be compatible with the utilization of a fixed amount of propellant over a fixed thrust time for any example mission. These values are 3660 kg of mercury, including 10 percent excess to accommodate mission

uncertainties and 666 days thrust time for the example Comet Halley rendezvous mission selected in paragraph 3.2.

3.5.2 THRUSTER ARRAY DEFINITION

The thruster array for the baseline spacecraft designs is composed of 30 mercury electron bombardment ion engines arranged in a 3 x 10 matrix as shown on Figure 3-5. This design provides for 24 operating engines and 6 spares. Sixteen of the engines are gimbaled, four on each end and four on each side. This configuration represents the final of several iterations during which various numbers of ion engines and their arrangement were investigated. The following paragraphs present the final ion engine sizing for the array presented on Figure 3-5.

The weight of a single thruster is given by the relation (Reference 3-6)

$$m_t = 1.85 + 57\, D_b^{\,2} \quad \text{kg}$$

where m_t = individual thruster mass, kg

D_b = ion beam diameter, m

The beam diameter based on 32 A/m^2, is given by

$$D_b = \frac{630}{I_{sp}} \left(\eta_m \eta_t \, P_t\right)^{\frac{1}{2}} \quad \text{m}$$

where I_{sp} = time specific impulse, seconds

= 6000 seconds for this example mission

η_m = propellant utilization efficiency

= 0.90

η_t = total thruster efficiency

= $\eta_m \, \eta_p$

η_p = power efficiency

= 0.927 (estimated from Reference 3-1)

P_t = maximum thruster power

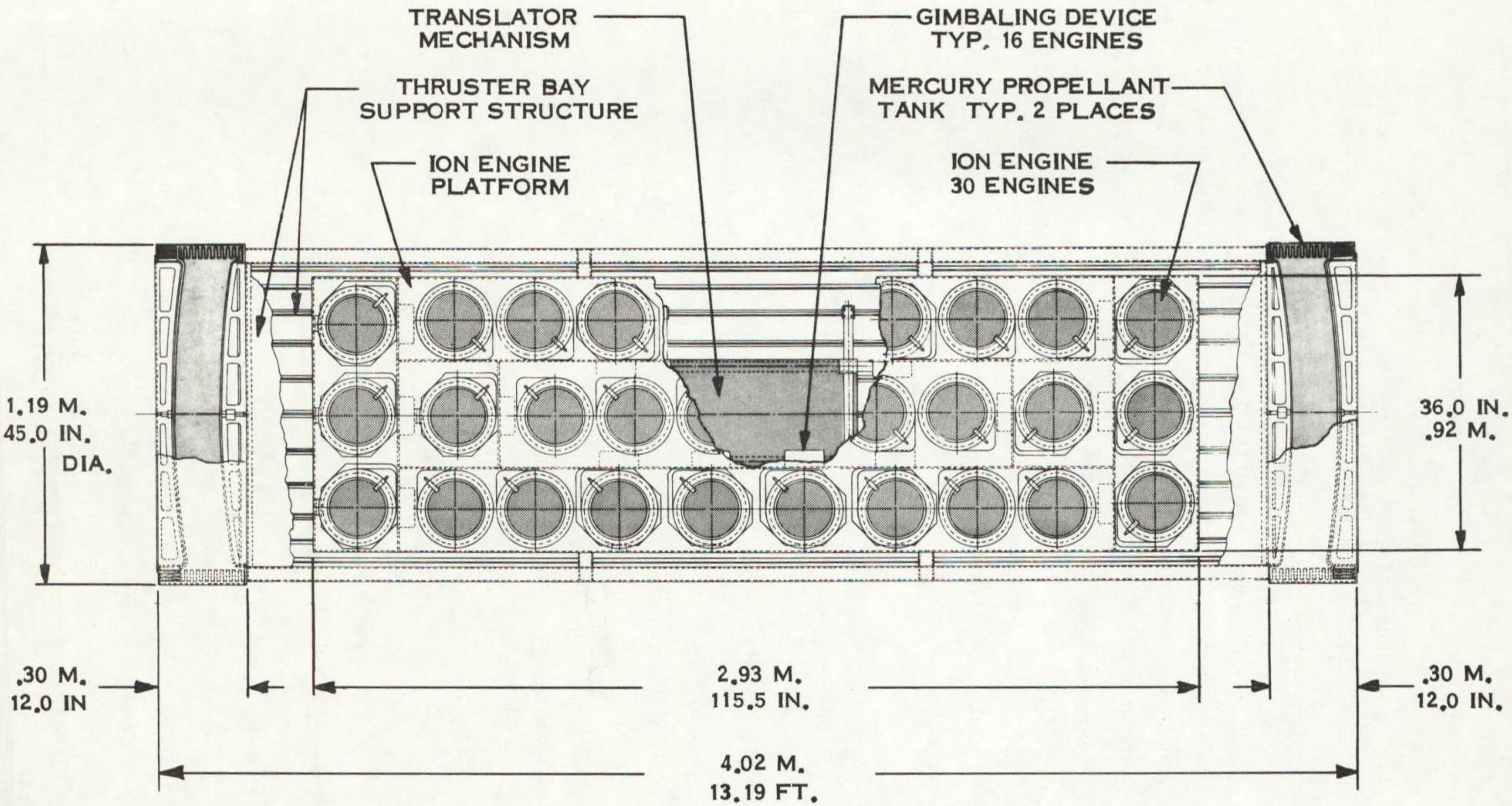
The maximum thruster power can be calculated from the relation

$$\dot{M}_p = 0.753 \times 10^{-2} \, \frac{\eta_p \, P_b}{\eta_m \, V_b} \quad \text{kg/hr}$$

FIGURE 3-5

BASELINE THRUSTER ARRAY

120 Kwe COMET HALLEY RENDEZVOUS
THERMIONIC SPACECRAFT



where $\dot{M}_p$ = propellant flow rate per ion engine

V_b = grid voltage, kv

The grid voltage is about 4 kV for the example 6000 second specific impulse required for the example mission. Based on the actual propellant required to complete the example mission (Table 3-2), and the 666 days of thruster operation, the total proepllant flow rate is 0.208 kg/hr, or an $\dot{M}_p$ of 0.866 x 10^{-2} kg/hr for each of the 24 operating engines. Therefore, for η_m of 0.90 and η_p of 0.927, the maximum power for each thruster is calculated at 4.46 kWe. The beam diameter, D_b, is then calculated at 0.193 meters. The actual engine diameter is 30 percent greater than the beam diameter or about 0.25 meters.

The ion engine unit weight is than calculated at about 4.0 kg. The total weight of all 30 ion engines is 120 kg.

The thruster array component weights are summarized on Table 3-4. These have been estimated using both the relations given in Reference 3-6, and the actual thruster array layout presented on Figure 3-5. Beryllium structure is assumed.

It is noted that the 24 operating ion engines require a total of 107 kWe delivered to the thruster array. As presented in Section 4.0 below, this requirement is slightly exceeded by the 40 VDC external fuel and flashlight thermionic reactor powerplants, which provide about 109 kWe. Due to less efficient power conditioning, the 10 VDC flashlight reactor spacecraft delivers only 102 kWe to the thruster array.

3.6 PROPELLANT SYSTEM

The propellant system consists of the mercury propellant, its containment tanks, and the propellant distribution system. The 3660 kg of mercury propellant required for this example mission, including a 10 percent allowance for mission uncertainties, is located in two stainless steel tanks, on each side of the thrust array as shown on Figure 3-5. This arrangement is required for the side thrust spacecraft to assure center of gravity control throughout the mission. This propellant system is common to all the baseline spacecraft designs investigated in this study.

TABLE 3-4

THRUST SUBSYSTEM MASS SUMMARY
120 kWe NUCLEAR THERMIONIC SPACECRAFT

COMPONENT	ELEMENT (KG)	TOTAL (KG)
THRUSTER* (30)	4.0	120.0
THRUST VECTOR CONTROL		
GIMBAL ACTIVATORS (16)	1.8	28.8
TRANSLATOR ACTIVATORS (2)	2.7	5.4
ELECTRONICS	9.0	9.0
MOUNTING STRUCTURE	19.4	19.4
TRANSLATOR BEARINGS	2.0	2.0
TRANSLATOR SUPPORTS	2.0	2.0
ACTIVATOR CARRIAGE	4.0	4.0
MISCELLANEOUS (WIRING, LAUNCH CAGING ADAPTORS, ETC.)	22.4	22.4
CONTROLLER**	5.0	5.0
TOTAL		213.0

*24 OPERATING; 6 SPARES
**LOCATED IN PAYLOAD BAY, NOT INCLUDED IN TOTAL WEIGHT

The example tank design illustrated provides for positive mercury expulsion via a metal bellows system pressurized by a cold gas system. This also assures that no voids will form in the tanks during the mission coast phase, which, if incurred, would result in radiation streaming.

The mass of the propellant storage tanks and the propellant feed system is estimated at three percent of the propellant mass, or 110 kg. The details of the propellant feed system have not been investigated. This feed system must assure that the flow from the two propellant tanks is uniform to the point that the spacecraft is not unbalanced beyond the compensating capability of the TVC system. Further study is required in this area. It is certain that metering accuracy of the propellant feed system, and the axial translation requirements of the TVC will be closely related.

4.0 SPACECRAFT DESIGN SUMMARY

This section summarizes the performance of the External Fuel and Flashlight reactor spacecrafts designed for the Comet Halley rendezvous mission. The baseline power level is 120 kWe, P_e, to the thrust subsystem. Study guidelines, launch vehicle details, and common spacecraft subsystems are presented in Section 3.0. Details of the external fuel reactor spacecraft is presented in Volume II, and the external fuel (Flashlight) reactor spacecraft is presented in Volume III.

4.1 EXTERNAL FUEL REACTOR SPACECRAFT

A detailed design layout, weight summary, and power balance and distribution are presented for the two baseline designs, the Independently Pumped Diode (IPD) reactor spacecraft and the Heat Pipe Cooled Diode (HCD) reactor spacecraft. Each of the baseline designs are perturbed to show the effect of:

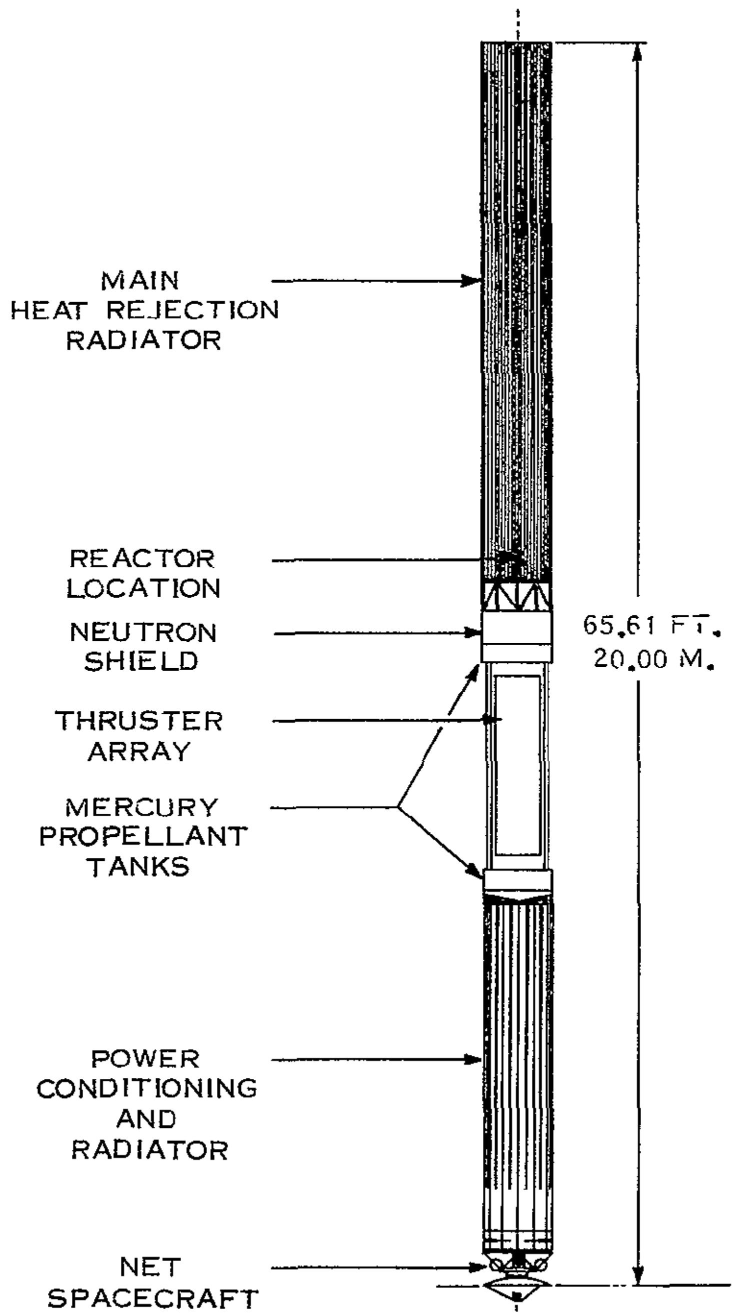
- Launch by the Advanced Logistics System (ALS)
- Replacing U-235 fuel with U-233 fuel in the reactor diodes
- Use of DC EM pumps
- Replacing the single coolant loop heat rejection system with one of multiple, independent loops.

4.1.1 IPD REACTOR SPACECRAFT BASELINE DESIGNS

This section describes the external fuel reactor spacecraft powered by the IPD reactor in which each of the diodes are independently cooled by pumped NaK-78. The basic configuration of the baseline design is shown in Figure 4.1. The spacecraft is 20 m long with a 1 m payload extension boom that is required to satisfy coincidence of the center-of-gravity with center-of-thrust. The propulsion system specific weight is 29.6 kg/kWe at 120 kWe, P_e. The 1.14 m diameter spacecraft locates all high temperature components (1033^{o}K), the stainless steel tube main radiator and reactor, at the forward end, and locates all lower temperature components (367^{o}K), the power conditioning units and net payload, at the aft end of the spacecraft. The thruster bay of 30 ion engines, 24 of which are operating at any one time, and mercury propellant tanks on either side of the thruster bay are located in the center of the spacecraft. Advantages of the side-thrust configuration are the separation of high and low temperature components, the nearly 4 π steradian

FIGURE 4-1

INDEPENDENTLY PUMPED DIODE SPACECRAFT BASELINE DESIGN
120 KWE COMET HALLEY RENDEZVOUS SPACECRAFT



viewing angle available for the payload, and the potential capability of the spacecraft to rotate about the thrust vector without compromising the effective thrust.

Weight summary for the baseline IPD reactor spacecraft design is presented in Table 4-1. The launch vehicle payload requirement is 8690 kg, which includes 706 kg of flight shroud weight penalty. Weight of each of the remaining major spacecraft systems is:

• Propulsion System	3552 kg
• Propellant System	3770 kg
• Net Payload	662 kg

The propellant requirements and net payload have been obtained from mission analysis of the example 940-day Comet Halley rendezvous mission (paragraph 3.2).

The power balance and distribution for the baseline IPD reactor spacecraft is shown in Figure 4-2. The electrical requirements of the spacecraft are based on the 120 kWe power level, which is input to the thrust system. Of the 120 kWe, about 95 percent supplies power to the high voltage ion engine screen grid and the remaining 5 percent is utilized by miscellaneous ion engine loads. For all the 120 kWe spacecraft designs 1 kWe each is allocated for the payload and the powerplant and spacecraft control. An additional 0.7 kWe powers the reactor control system. For the IPD reactor spacecraft, 7.5 kWe is required by the DC EM pump to circulate NaK-78 throughout the heat rejection system, and 5.5 kw_t is dissipated by the low voltage cable as I^2R losses. Therefore, a total of 135.7 kWe of reactor output power is required to supply 120 kWe to the thrust subsystem and subsequently, 100 kWe to beam power.

4.1.2 HCD REACTOR SPACECRAFT BASELINE DESIGN

The basic configuration of the HCD reactor spacecraft, shown in Figure 4-3, is identical to that of the IPD reactor spacecraft except that a beryllium/stainless steel tube and fin main radiator has been used, and AC pumps circulate the primary coolant. The weight summary for the baseline HCD reactor spacecraft is presented in Table 4-2. The propulsion system specific weight is 27.2 kg/kWe at 120 kWe. For this spacecraft, the launch vehicle payload requirement is 8411 kg,

TABLE 4-1

WEIGHT SUMMARY
BASELINE IPD EXTERNAL FUEL SPACECRAFT

COMPONENTS	WEIGHT, KG		
PROPULSION SYSTEM			3552
POWER SYSTEM		2714	
REACTOR	1410		
HEAT REJECTION	677		
NEUTRON SHIELD	519		
PUMP LOW VOLTAGE CABLE	48		
STRUCTURE	60		
THRUST SYSTEM		838	
THRUST ARRAY	213		
POWER CONDITIONING	306		
POWER CONDITIONING RADIATOR	96		
LOW VOLTAGE CABLE	140		
HIGH VOLTAGE CABLE	3		
STRUCTURE	80		
PROPELLANT SYSTEM			3770
PROPELLANT		3660	
TANKS AND DISTRIBUTION		110	
NET SPACECRAFT			662
FLIGHT SHROUD WEIGHT PENALTY			706
LAUNCH VEHICLE PAYLOAD REQUIREMENT			8690

FIGURE 4-2

EXTERNAL FUEL INDEPENDENTLY PUMPED DIODE SPACECRAFT - 120 kWe
POWER BALANCE AND DISTRIBUTION

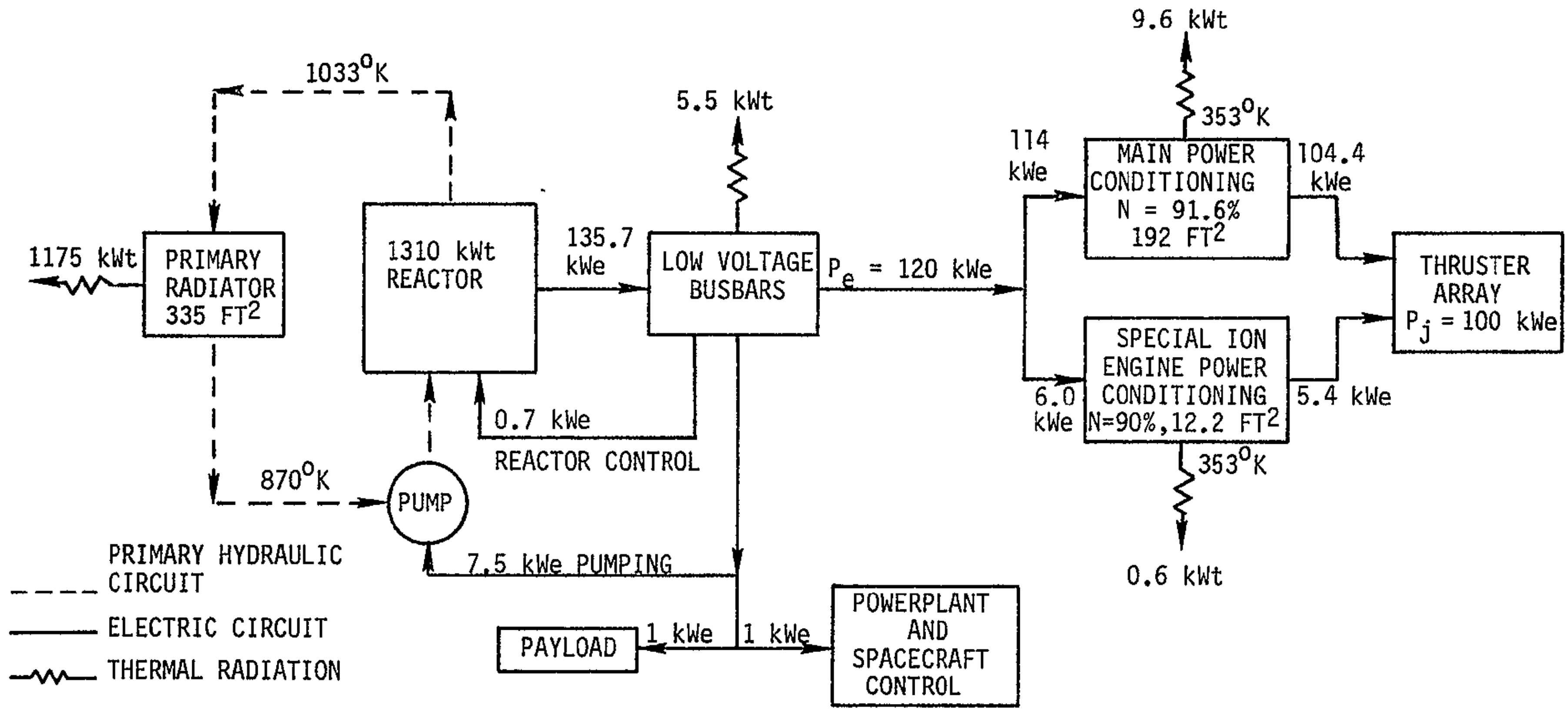


FIGURE 4-3

HEAT PIPE COOLED DIODE SPACECRAFT BASELINE DESIGN
120 KWE COMET HALLEY RENDEZVOUS SPACECRAFT

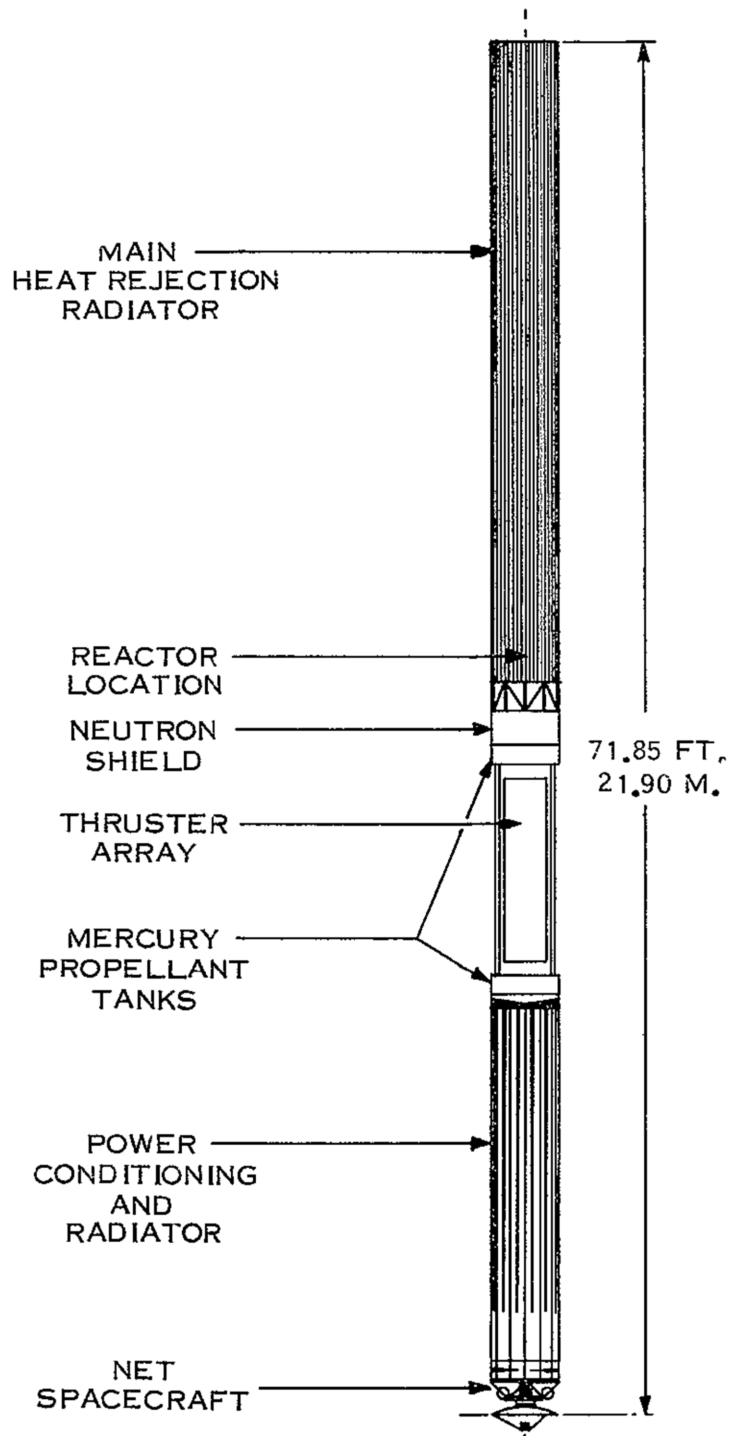


TABLE 4-2

WEIGHT SUMMARY
BASELINE HCD EXTERNAL FUEL SPACECRAFT

COMPONENTS	WEIGHT, KG		
PROPULSION SYSTEM			3322
POWER SYSTEM		2490	
REACTOR	1390		
HEAT REJECTION	512		
NEUTRON SHIELD	519		
HOTEL POWER CONDITIONING	14		
HOTEL POWER CONDITIONING RADIATOR	5		
PUMP LOW VOLTAGE CABLE	2		
STRUCTURE	48		
THRUST SYSTEM		832	
THRUST ARRAY	213		
POWER CONDITIONING	306		
POWER CONDITIONING RADIATOR	96		
LOW VOLTAGE CABLE	134		
HIGH VOLTAGE CABLE	3		
STRUCTURE	80		
PROPELLANT SYSTEM			3770
PROPELLANT		3660	
TANKS AND DISTRIBUTION		110	
NET SPACECRAFT			662
FLIGHT SHROUD WEIGHT PENALTY			657
LAUNCH VEHICLE PAYLOAD REQUIREMENT			8411

657 kg of which is flight shroud weight penalty. The remaining major spacecraft system weights include:

- Propulsion System 3322 kg
- Propellant System 3770 kg
- Net Payload 662 kg

The power balance and distribution for the HCD reactor spacecraft is shown in Figure 4-4. The electrical requirements of this spacecraft are also based on 120 kWe input to the thrust subsystem. The power requirements are similar to those discussed for the IPD reactor spacecraft. The only differences are that only 2.9 kWe are required by the AC pumps and 5.1 kw_t of power are dissipated as I^2R losses from the low voltage cable. Consequently, 130.7 kWe of reactor output power are required to supply 120 kWe to the thrust system and 100 kWe to the thruster beam power.

4.1.3 ALTERNATE EXTERNAL FUEL REACTOR SPACECRAFT DESIGNS

The baseline IPD reactor spacecraft design and the baseline HCD reactor spacecraft, have been perturbed to show the effect of:

- Launch by Advanced Logistics System (ALS)
- Use of U-233 Fueled Reactors
- AC vs. DC EM Pumps
- Multiple Radiator Loops

4.1.3.1 ALS-Launched External Fuel Reactor Spacecraft Designs

The IPD reactor spacecraft and HCD reactor spacecraft baseline designs have been reconfigured for launch by the ALS. A comparison of significant parameters between the baseline and alternate designs of the external fuel reactor spacecraft is presented in Table 4-3.

The primary spacecraft constraint introduced by the ALS is that the launch vehicle payload bay is limited to 18.3 m in length. Although folding the spacecraft may be an acceptable method of shortening it for ALS launch, further study is required. This option is not permitted by the study ground rules and constraints. Therefore, reducing the spacecraft length was accomplished by increasing the spacecraft diameter to meet heat rejection area requirements. Since the 1.14 m diameter of the baseline spacecraft represents the maximum value for

FIGURE 4-4

EXTERNAL FUEL HEAT PIPE COOLED DIODE SPACECRAFT - 120 kWe POWER BALANCE AND DISTRIBUTION

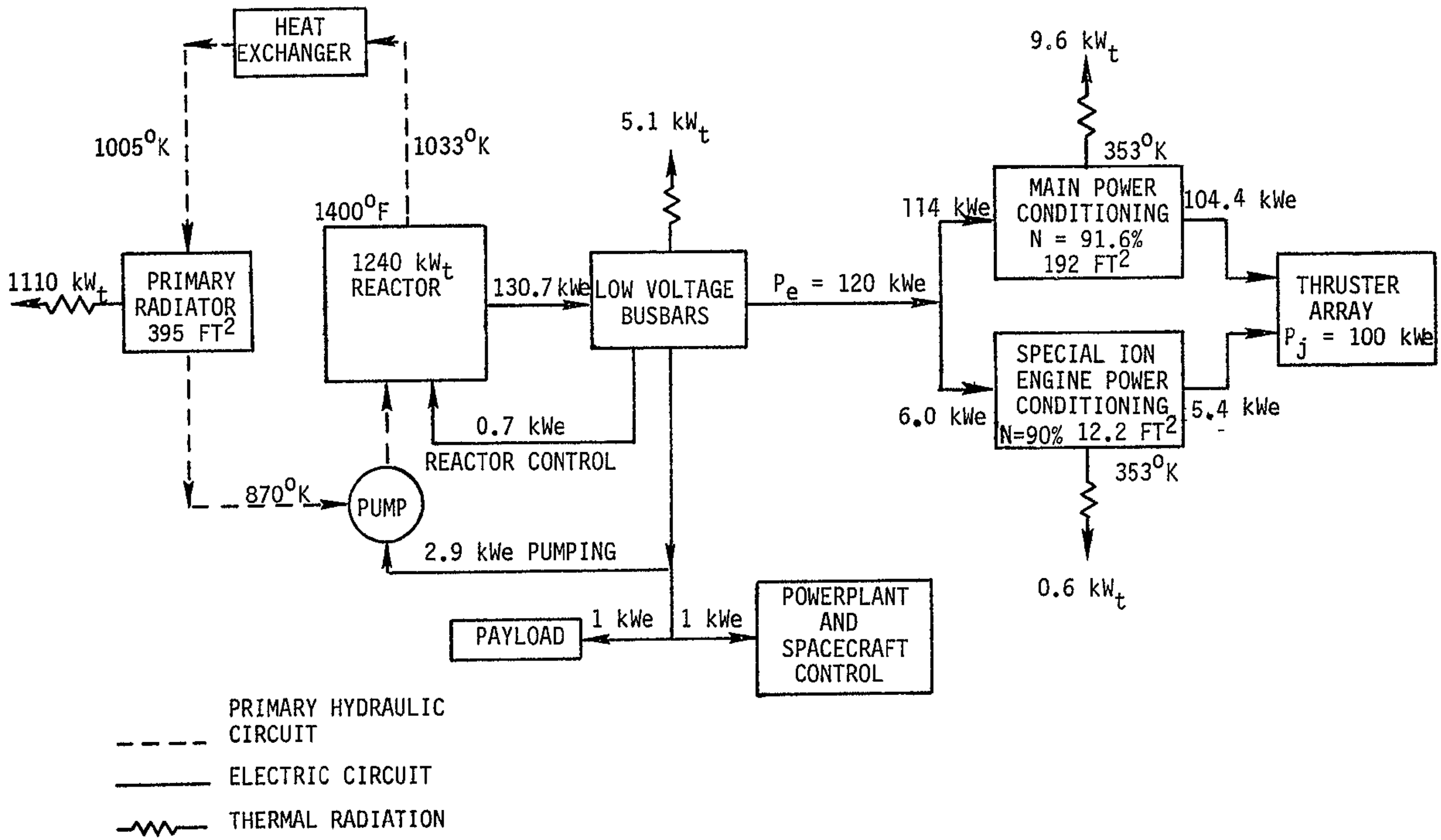


TABLE 4-3

COMPARISON OF BASELINE AND ALS-LAUNCHED DESIGNS
FOR EXTERNAL FUEL REACTOR SPACECRAFT

PARAMETER	HEAT PIPE COOLED REACTOR		MULTI-DUCTED EM PUMP COOLED REACTOR	
	BASELINE	ALS-LAUNCHED	BASELINE	ALS-LAUNCHED
Propulsion Subsystem Specific Weight, kg/kWe	27.7	29.4	29.6	31.6
Propulsion Subsystem Weight, kg	3322	3525	3525	3797
Length, m	21.9	18.3	21.0	18.3
Diameter, m	1.14	1.43	1.14	1.31
Launch Weight, kg	8411	7957	8690	8228
Ion Engine Input Power, kWe	109.7	109.7	109.7	109.7
Gross Reactor Output Power, kWe	130.7	130.4	135.7	132.7
Reactor Thermal Power, kW	1240	1235	1310	1285
Payload at Spacecraft End, Percent	100	100 (Boomed 1.0m)	100 (Boomed 1.0m)	100 (Boomed 1.0m)

which the mercury propellant provides adequate gamma shielding, an increase in diameter requires an increase in neutron shielding and the addition of tungsten permanent gamma shielding.

Consequently, for the IPD reactor spacecraft the spacecraft diameter was increased to 1.31 m, which results in an increase in propulsion system specific weight of 2 kg/kWe. The diameter of the baseline HCD reactor spacecraft is increased to 1.43 m with a resultant increase in propulsion system specific weight of 1.7 kg/kWe. The launch weight indicated in Table 4-3 includes no launch shroud weight penalty, which is not required for the ALS.

The spacecraft center of gravity must be coincident with the center of thrust. Meeting this requirement can be somewhat more difficult in the side thrust configuration, where the thrust vector is perpendicular to the axis of the vehicle, relative to an end thrust spacecraft, where the thrust vector is parallel to the axis of the vehicle. This study assumes that the net spacecraft can be relocated if necessary to meet the center of gravity requirement. The total concentrated mass of the net spacecraft may be boomed away from the power conditioning radiator when it is necessary to shift the center of gravity in that direction to match the center of gravity with the center of thrust. Alternately it is assumed that up to 95 percent of the net spacecraft can be moved to immediately behind the last mercury propellant tank when necessary to shift the center of gravity from a location between the center of the thruster array and the power conditioning to exactly the center of the thruster array. All of the net spacecraft cannot be located in this manner since the antenna and the basic sensors must remain at the end of the spacecraft. This approach, although sometimes required in this study, is very undesirable, since the net spacecraft must be configured in two separate units.

For the IPD reactor spacecraft in the ALS configuration, the payload boom must be extended from 1 m to 2.1 m. Similarly, the HCD reactor spacecraft must extend the net payload 1 m; whereas, the baseline design requires no payload relocation.

4.1.3.2 U-233 Fueled External Fuel Reactor Spacecraft

Alternate external fuel reactor spacecraft designs were generated by replacing the baseline U-235 fueled reactor diodes with those fueled with U-233. Comparison of the U-233 fueled reactor spacecraft design and the baseline spacecraft design is made in Table 4-4. The only change in the spacecraft is a decrease in reactor size and weight. The IPD reactor with U-233 fuel weighs 485 kg less and results in a decrease in propulsion system specific weight of 4.1 kg/kWe. The HCD reactor using U-233 fuel weighs 480 kg less than the baseline U-235 fueled reactor and results in a decrease of 4 kg/kWe in propulsion system specific weight. As indicated in Table 4-4, the decrease in weight in the forward end of the spacecraft necessitates net payload relocation to the forward payload bay, which is located between the aft mercury propellant tank and the power conditioning section.

4.1.3.3 Use of DC EM Pumps in External Fuel Reactor Spacecraft

This paragraph discusses the effect on spacecraft design of replacing the baseline AC pumps with DC pumps to circulate NaK-78 through the primary heat rejection system. Only the HCD reactor spacecraft will be considered since the baseline IPD reactor spacecraft uses a multi-ducted DC EM pump.

Table 4-5 compares the significant spacecraft design parameters for the baseline and DC pump alternate HCD reactor spacecraft design. The use of DC pumps results in an increase in propulsion system specific weight of 0.6 kg/kWe and spacecraft length from 21.9 m to 23.5 m. Moreover, reactor output power increases from 130.7 kWe to 135.3 kWe. The increase in propulsion system weight and power requirement is due primarily to the less efficient power conditioning and the greater low voltage cable loss associated with DC pumps. Also, since DC pumps are lighter than AC pumps and the alternate design requires more power conditioning weight, 10 percent of the net payload must be relocated to the forward payload bay in order that the center-of-gravity constraint be satisfied.

4.1.3.4 Multiple Radiator Loop External Fuel Reactor Spacecraft Design

The final alternate spacecraft design has been generated by replacing the single loop main radiator with a multiple loop heat rejection

TABLE 4-4

COMPARISON OF BASELINE AND U-233 FUELED FOR EXTERNAL FUEL REACTOR SPACECRAFT

PARAMETER	HEAT PIPE COOLED REACTOR		MULTI-DUCTED EM PUMP COOLED REACTOR	
	BASELINE	U-233 FUEL	BASELINE	U-233 FUEL
Propulsion Subsystem Specific Weight, kg/kWe	27.7	23.7	29.6	25.5
Propulsion Subsystem Weight, kg	3322	2842	3552	3067
Length, m	21.9	20.9	21.0	20.
Diameter	1.14	1.14	1.14	1.14
Launch Weight, kg	8411	7892	8690	8205
Ion Engine Input Power, KWe	109.7	109.7	109.7	109.7
Gross Reactor Output Power, kWe	130.7	130.7	135.7	135.7
Reactor Thermal Power, kW	1240	1240	1310	1310
Payload At Spacecraft End, Percent	100	70	100 (Boomed 1.0m)	80

TABLE 4-5

COMPARISON OF BASELINE AND DC EM PUMP
SPACECRAFT DESIGN FOR EXTERNAL FUEL REACTOR SPACECRAFT

PARAMETER	HEAT PIPE COOLED REACTOR	
	BASELINE	DC EM PUMP
Propulsion Subsystem Specific Weight, kg/kWe	27.7	28.3
Propulsion Subsystem Weight, kg	3322	3391
Length, m	21.9	23.5
Diameter, m	1.14	1.14
Launch Weight, kg	8411	8553
Ion Engine Input Power, kWe	109.7	109.7
Gross Reactor Output Power, kWe	130.7	135.3
Reactor Thermal Power, kw	1240	1280
Payload at Spacecraft End, Percent	100	90

system in the HCD reactor spacecraft. This perturbation to the baseline IPD reactor spacecraft design was not considered because the IPD concept already has greater multiplicity with a separate radiator loop for each diode.

The single loop radiator in the HCD reactor spacecraft was replaced by a radiator of four independent loops, one of which is redundant. Comparison between the baseline and alternate designs is shown in Table 4-6. The increased weight and area of the alternate radiator yielded an increase in propulsion system specific weight of 0.3 kg/kWe and spacecraft length from 21.9 m to 22.9 m. To compensate for the increased weight at the forward end of the spacecraft, the net payload has to be boomed 0.3 m from the spacecraft.

4.2 FLASHLIGHT REACTOR SPACECRAFT

An overall layout, weight summary and power balance are given for two baseline designs; the first uses a reactor design with a 10-volt power output while the second assumes the 10-volt reactor design can be designed with a 40-volt power output with no penalty. Alternates to each of the baseline designs are presented with the variations being:

- Launch by an Advanced Logistics System (ALS)
- Use of a U-233 fueled reactor
- Use of DC powered EM pumps in the heat rejection subsystem
- Use of multiple and redundant primary radiator loops.

4.2.1 BASELINE 10 VDC FLASHLIGHT REACTOR SPACECRAFT

The general layout and configuration of the Baseline 10-volt Flashlight Reactor Spacecraft, shown on Figure 4-5, is similar to the Externally Fueled Reactor Spacecraft. The primary difference is in the length of the primary radiator and the power conditioning radiator. The overall spacecraft length is 27. 1 m long and its diameter is 1.14 m. The high temperature primary radiator of tube and fin configuration occupies the forward end of the spacecraft. The thruster bay of 30 ion engines and two mercury propellant tanks comprises the center section of the spacecraft and is identical in size and construction with the thruster bays of the Externally Fueled Reactor Spacecraft. A 39 m long power conditioning and radiator bay makes up most of the

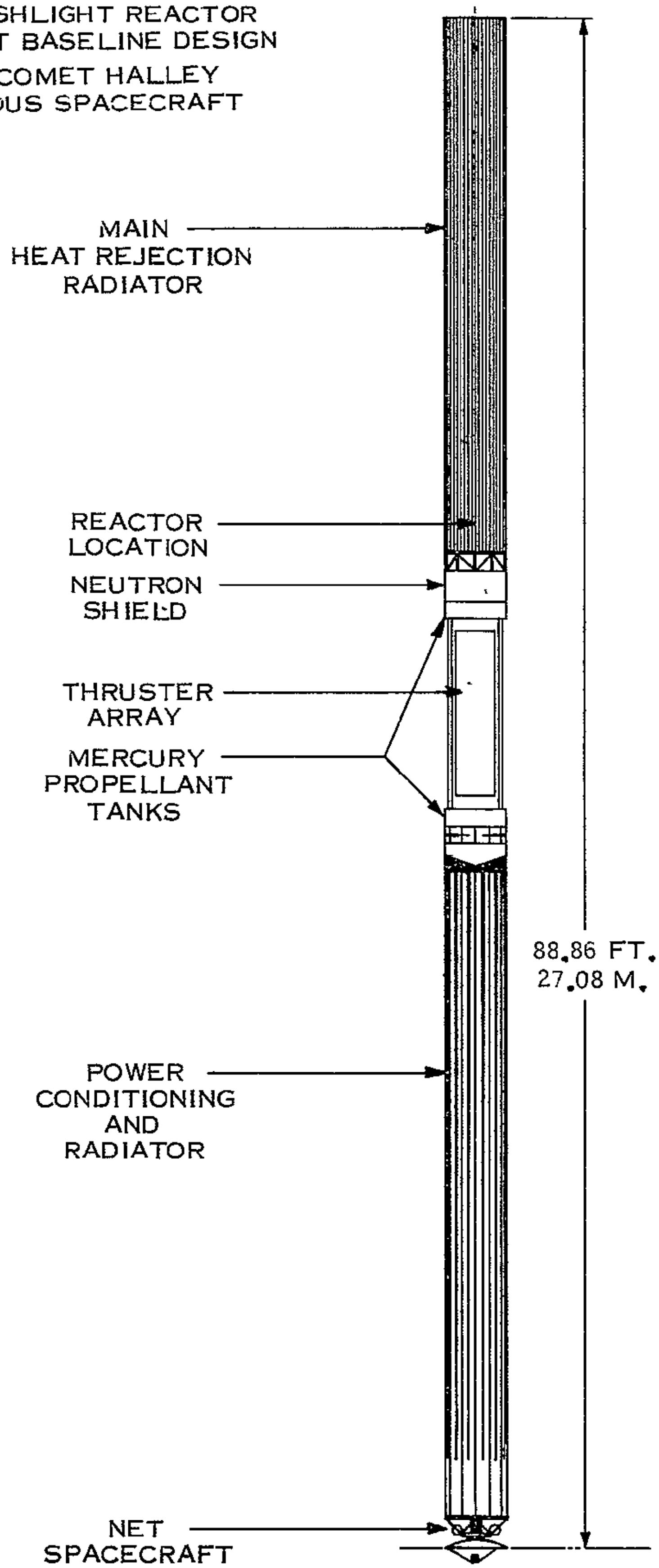
TABLE 4-6

COMPARISON OF BASELINE AND MULTIPLE RADIATOR
SPACECRAFT DESIGNS FOR EXTERNAL FUEL REACTOR SPACECRAFT

PARAMETER	HEAT PIPE COOLED REACTOR	
	BASELINE	MULTIPLE RADIATOR LOOPS
Propulsion Subsystem Specific Weight, kg/kWe	27.7	28.0
Propulsion Subsystem Weight, kg	3322	3359
Length, m	21.9	22.9
Diameter, m	1.14	1.14
Launch Weight, kg	8411	8499
Ion Engine Input Power, kWe	109.7	109.7
Gross Reactor Output Power, kWe	130.7	131
Reactor Thermal Power, kw	1240	1242
Payload at Spacecraft End, Percent	100	100 (Boomed 0.3m)

FIGURE 4-5

10 VDC FLASHLIGHT REACTOR
SPACECRAFT BASELINE DESIGN
120 KWE COMET HALLEY
RENDEZVOUS SPACECRAFT



remainder of the spacecraft. A payload bay with sensors for scientific experiments and antennae for communications is located at the end of the spacecraft.

A weight summary of the 10-volt Baseline Flashlight Reactor Spacecraft is listed on Table 4-7. The overall launch vehicle payload requirement of 9280 kg consists of the following major system weights:

- Propulsion System 3880 kg
- Propellant System 3770 kg
- Net Spacecraft (Payload) 662 kg
- Flight Shroud Weight Penalty 968 kg

The Propellant System and Net Spacecraft weights are identical with all the other spacecraft designs. The Propulsion System Specific Weight is 32.3 kg/kWe, at 120 kWe, P_e.

Figure 4-6 presents the power balance of the 10-volt Baseline design. The requirements of 120 kWe to the Thruster Subsystem and 2 kWe for the payload and the powerplant and spacecraft control function is identical with those of the other spacecraft designs. A reactor power output of 167 kWe is required due to large losses (36.4 kWe) in the low voltage cable. High losses in the main power conditioning because of the 10 VDC input power reduce the beam power from the ion engines to 92.7 kWe in this design.

4.2.2 BASELINE 40 VDC FLASHLIGHT REACTOR SPACECRAFT

The overall configuration of the 40-volt Baseline Flashlight Reactor spacecraft is identical in component layout to the 10-volt system. The major changes in the 40-volt system are in the much lighter power transmission lines and the much lower electrical line losses, which allows a lower reactor power level and a corresponding smaller heat rejection system.

The general arrangement of the spacecraft components are shown on Figure 4-7. The spacecraft diameter is 1.14 meters as in the 10-volt system but the overall spacecraft length is only 20.75 meters, with the short length attributable entirely to the comparatively small primary and PC radiators. The primary radiator is 8.4 m long and located at the front end of the spacecraft. The thruster bay and propellant tanks are identical in design to the 10-volt system component.

TABLE 4-7

WEIGHT SUMMARY
BASELINE 10-VOLT FLASHLIGHT REACTOR SPACECRAFT

COMPONENTS	WEIGHT, KG		
PROPULSION SYSTEM			3880
POWER SUBSYSTEM		2295	
REACTOR	1062		
HEAT REJECTION	622		
NEUTRON SHIELD	531		
HOTEL POWER CONDITIONING	25		
HOTEL POWER CONDITIONING RADIATOR	8		
PUMP LOW VOLTAGE CABLE	1		
STRUCTURE	46		
THRUST SUBSYSTEM		1585	
THRUSTER ARRAY	213		
POWER CONDITIONING	357		
POWER CONDITIONING RADIATOR	188		
LOW VOLTAGE CABLE	722		
HIGH VOLTAGE CABLE	3		
STRUCTURE	102		
PROPELLANT SYSTEM			3770
PROPELLANT		3660	
TANKS AND DISTRIBUTION		110	
NET SPACECRAFT			662
FLIGHT SHROUD WEIGHT PENALTY			968
LAUNCH VEHICLE PAYLOAD REQUIREMENT			9280

FIGURE 4-6

10 VOLT FLASHLIGHT REACTOR SPACECRAFT BASELINE DESIGN - 120 kWe POWER BALANCE AND DISTRIBUTION

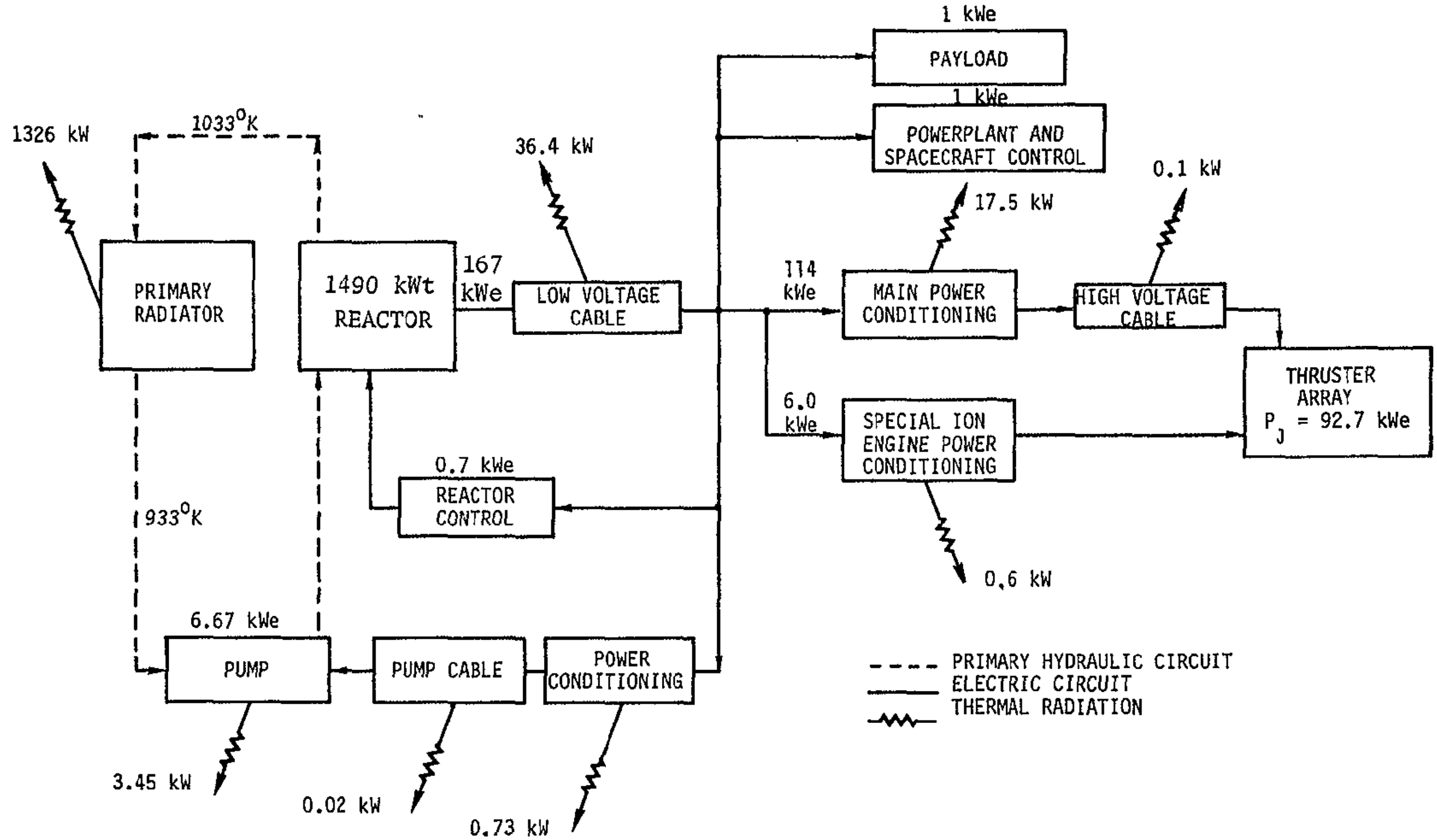
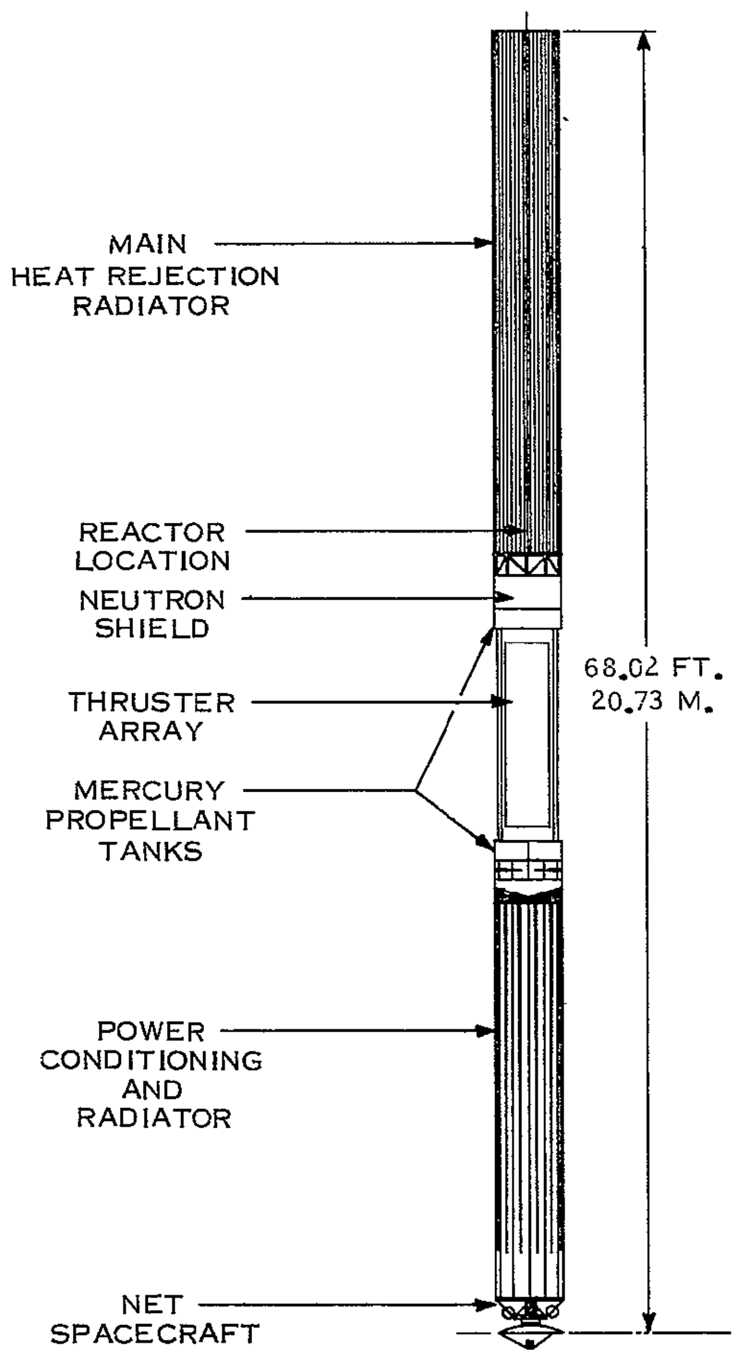


FIGURE 4-7

40 VDC FLASHLIGHT REACTOR SPACECRAFT BASELINE DESIGN
120 KWE COMET HALLEY RENDEZVOUS SPACECRAFT



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The power conditioning radiator is 12-sided in cross section as in the 10-volt design, but only 6.7 m in length. This shorter length, as compared to the 10-volt PC radiator, reflects the higher power conditioning efficiency for the 40-volt system and the lower reactor power output.

The detailed weight summary of the 40-volt Baseline Flashlight Reactor Spacecraft is given in Table 4-8. The major system weights are as follows:

• Propulsion System	3066 kg
• Propellant System	3770 kg
• Net Spacecraft	662 kg
• Flight Shroud Weight Penalty	690 kg

The total spacecraft launch weight is 8188 kg and the propulsion system specific weight is 25.55 kg per kWe of net power to the thrust subsystem. The total net power, P_e, is 120 kWe as in the other baseline spacecraft, of which about 100 kWe eventually appears as beam power from the operating ion engines, as shown in the power balance diagram of Figure 4-8. The required reactor output power is 136 kWe and its thermal power is 1310 kWt. Only 16 kWe is needed to supply the powerplant and payload requirements, and the losses in the electrical network. This non-propulsive power requirement is only one-third of the corresponding requirement in the 10-volt system, with the difference attributable to the much lower electrical losses in the low voltage power transmission cables.

4.2.3 ALTERNATE FLASHLIGHT REACTOR SPACECRAFT DESIGNS

4.2.3.1 ALS-Launched Flashlight Reactor Spacecraft

The ALS launched spacecraft are shortened in length to fit the 18.3 m length of the ALS cargo bay. The length reduction is accomplished by increasing the spacecraft diameter. Table 4-9 compares significant parameters in the 10-volt and 40-volt Baseline designs with their corresponding parameters in the ALS launched configurations. In the 10-volt systems, the ALS design propulsion system is 830 kg heavier due entirely to the heavier shield required in the larger diameter spacecraft which increases to 1.72 m. The propulsion system specific weight increases almost 7 kg/kWe to 39.2 kg/kWe. Although the propulsion system weight increases significantly, the launch weight actually decreases since the flight shroud weight penalty is eliminated for the ALS design.

TABLE 4-8

WEIGHT SUMMARY
BASELINE 40-VOLT FLASHLIGHT REACTOR SPACECRAFT

COMPONENTS	WEIGHT, KG		
PROPULSION SYSTEM			3066
POWER SYSTEM		2197	
REACTOR	1062		
HEAT REJECTION	535		
NEUTRON SHIELD	520		
HOTEL POWER CONDITIONING	31		
HOTEL POWER CONDITIONING RADIATOR	5		
PUMP LOW VOLTAGE CABLE	1		
STRUCTURE	43		
THRUST SYSTEM		869	
THRUST ARRAY	213		
POWER CONDITIONING	310		
POWER CONDITIONING RADIATOR	105		
LOW VOLTAGE CABLE	154		
HIGH VOLTAGE CABLE	3		
STRUCTURE	84		
PROPELLANT SYSTEM			3770
PROPELLANT		3660	
TANKS AND DISTRIBUTION		110	
NET SPACECRAFT			662
FLIGHT SHROUD WEIGHT PENALTY			690
LAUNCH VEHICLE PAYLOAD REQUIREMENT			8188

FIGURE 4-8

BASELINE DESIGN - 40 V FLASHLIGHT REACTOR SPACECRAFT - 120 kWe
POWER BALANCE AND DISTRIBUTION

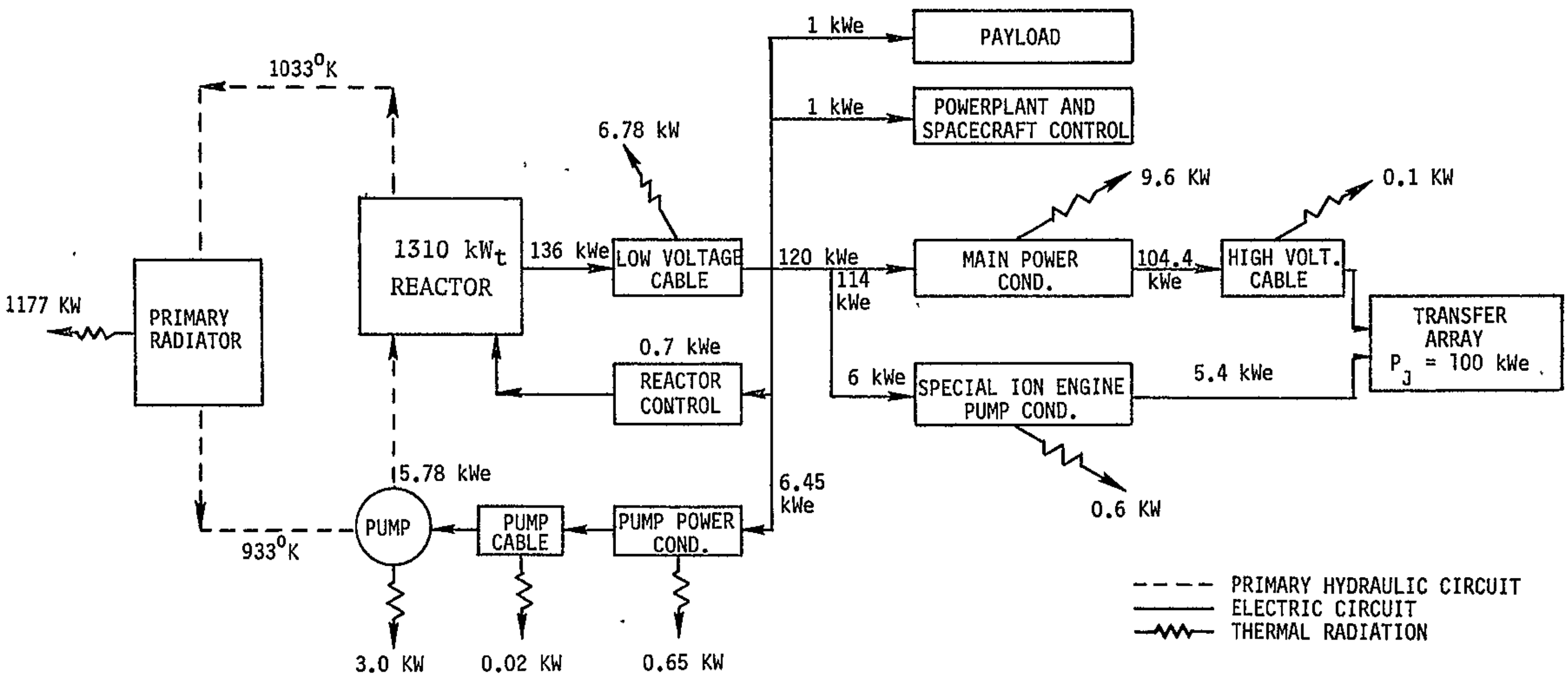

TABLE 4-9

COMPARISON OF BASELINE AND ALS-LAUNCHED DESIGNS
FOR FLASHLIGHT REACTOR SPACECRAFT

PARAMETER	10 VDC SYSTEMS		40 VDC SYSTEMS	
	BASELINE	ALS-LAUNCHED	BASELINE	ALS-LAUNCHED
Propulsion Subsystem Specific Weight, kg/kWe	32.3	39.2	25.6	28.0
Propulsion Subsystem Weight, kg	3879	4709	3066	3356
Length, m	27.1	18.3	20.75	17.5
Diameter	1.14	1.72	1.14	1.37
Launch Weight, kg	9279	9141	8188	7788
Ion Engine Input Power, kWe	101.84	101.84	109.7	109.7
Gross Reactor Output Power, kWe	166.8	155	136	134.7
Reactor Thermal Power, kw	1490	1418	1310	1302
Payload at Spacecraft End, Percent	12	95	80	100 (Boomed 1.8m)

In the 40-volt systems, ALS launch requires a diameter increase of 20 percent for the spacecraft to 1.37 meters and a propulsion system weight penalty increase of 2.4 kg/kWe to 28 kg/kWe. The launch weight decreases by 400 lbs. due to the elimination of the flight shroud.

4.2.3.2 U-233 Fueled Flashlight Reactor Spacecraft

Reactors fueled with U-233 are significantly smaller and lighter than U-235 fueled reactors in the Flashlight configuration. This results in lighter and smaller spacecraft but the magnitude of the decrease is different for the 10-volt and 40-volt reactor systems.

A comparison of the baseline designs with the U-233 fueled reactor alternates is presented in Table 4-10. In the 10-volt system, the U-233 reactor is about 325 kg lighter than the baseline reactor but only one-third of this savings is eventually realized in the propulsion system. A 0.6 m spacer section between the power system components and the thruster bay was needed to achieve the required spacecraft center of gravity. Additional low voltage cable weight and structure resulting from the spacer inclusion offset most of the weight savings in the reactor. The propulsion system decreases approximately 100 kg and the propulsion system specific weight decreases to 31.2 kg/kWe.

In the 40-volt systems, the U-233 reactor is almost 500 kg lighter than the baseline reactor and all of this savings is realized in the propulsion system weight. The net result is the smallest and lightest spacecraft determined in this study. The 40-volt U-233 fueled reactor design has a propulsion system specific weight of 20.8 kg/kWe and a launch weight of 7580 kg.

4.2.3.3 Flashlight Reactor Spacecraft with DC EM Pumps

Substitution of DC EM pumps for the AC pumps employed in the baseline designs results in similar penalties in both the 10-volt and 40-volt systems. In each case, the penalties are due to added weight and losses in the electrical system supplying the 1 volt power needed by the DC Pump. Key characteristics of the baseline and DC EM pump systems are summarized on Table 4-11 for the 10-volt and 40-volt designs.

TABLE 4 - 10
COMPARISON OF BASELINE AND U-233 FUELED REACTOR SPACECRAFT
FOR FLASHLIGHT REACTOR SPACECRAFT

PARAMETER	10 VDC SYSTEMS		40 VDC SYSTEMS	
	BASELINE	U-233 FUELED REACTOR	BASELINE	U-233 FUELED REACTOR
Propulsion Subsystem Specific Weight, kg/kWe	32.3	31.2	25.6	20.8
Propulsion Subsystem Weight, kg	3879	3760	3066	2488
Length, m	21.1	27.8	20.75	19.1
Diameter, m	1.14	1.14	1.14	1.14
Launch Weight, kg	9279	9162	8188	7580
Ion Engine Input Power, kWe	101.84	101.84	109.7	109.7
Gross Reactor Output Power, kWe	166.8	176.6	136	135.7
Reactor Thermal Power, kw	1490	1505	1310	1307
Payload at Spacecraft End, Percent	12	7	80	27

TABLE 4- 11

COMPARISON OF BASELINE AND DC EM PUMP SYSTEMS
FOR FLASHLIGHT REACTOR SPACECRAFT

PARAMETER	10 VDC SYSTEMS		40 VDC SYSTEMS	
	BASELINE	DC PUMP SYSTEM	BASELINE	DC PUMP SYSTEM
Propulsion Subsystem Specific Weight, kg/kWe	32.2	34.6	25.6	27.6
Propulsion Subsystem Weight, kg	3879	4150	3066	3310
Lenght, m	27.1	28.0	20.75	23.8
Diameter, m	1.14	1.14	1.14	1.14
Launch Weight, kg	9279	9602	8188	8572
Ion Engine Input Power,kWe	101.84	103.77	109.77	109.7
Gross Reactor Output Power kWe	166.8	176.7	136	146.6
Reactor Thermal Power,kw	1490	1553	1310	1370
Payload at Spacecraft End, Percent	12	7	80	50

Pump cable weights of 200-250 kg are necessary in the DC pump systems compared to 1 kg for the AC pump systems. Lower efficiencies in the pump power conditioning adds between 20 and 50 kg of increased weight to the DC systems. The net result is a propulsion system specific weight increase of about 2.3 kg/kWe, to 34.6 kg/kWe, for the 10-volt DC pump system with a corresponding increase in launch weight. The propulsion system specific weight increase for the 40-volt DC pump system is 2.0 kg/kWe to 27.6 kg/kWe.

4.2.3.4 Flashlight Reactor Spacecraft with Multiple Radiator Loops

Spacecraft employing four independent radiator loops, one of which is redundant, were investigated as alternates to the single radiator loop designs of the baseline systems. Table 4-12 presents a comparison of the significant parameters.

In both the 10-volt and 40-volt systems, the use of multiple radiator loops increase the primary radiator and the overall spacecraft lengths by about 10 meters. This relatively large increase is due to:

- The 33 percent area increase necessitated by the redundant loop.
- A 30°K radiator coolant inlet temperature drop due to the heat exchanger made necessary by the requirement of thermally connecting a single reactor loop with multiple radiator loops.
- A decrease in radiator average temperature due to reoptimization of the reactor temperature rise.

The additional pumps, piping, heat exchanger, etc., in the multiple radiator loop design results in heat rejection system weight increases of about 200 kg, which is reflected in similar increases in propulsion system weight and overall launch weight. In the 10-volt systems, multiple radiator loops increase the propulsion system specific weight by about 2 kg/kWe to 34.4 kg/kWe while the corresponding values in the 40-volt systems are 1.5 kg/kWe and 27.1 kg/kWe, respectively.

TABLE 4-12

COMPARISON OF BASELINE AND MULTIPLE RADIATOR LOOP DESIGNS
FOR FLASHLIGHT REACTOR SPACECRAFT

PARAMETER	10 VDC SYSTEMS		40 VDC SYSTEMS	
	BASELINE	MULTIPLE RADIATOR LOOPS	BASELINE	MULTIPLE RADIATOR LOOPS
Propulsion Subsystem Specific Weight, kg/kWe	32.3	34.3	25.6	27.1
Propulsion Subsystem Weight, kg	3879	4150	3066	3245
Length, m	27.1	37.2	20.75	31.9
Diameter, m	1.14	1.14	1.14	1.14
Launch Weight, kg	9279	9580	8188	8365
Ion Engine Input Power, kWe	101.84	101.84	109.7	109.7
Gross Reactor Output Power, kWe	166.8	167.3	136	133.5
Reactor Thermal Power, kW	1490	1492	1310	1300
Payload at Spacecraft End, Percent	12	63	80	100 (Extended 1.8m)

5.0 CONCLUSIONS

A. The 40 VDC internal fuel (Flashlight) and both 40 VDC external fuel spacecraft can meet Comet Halley Mission performance requirements.

B. The U-235 fueled 10 VDC internal fuel (Flashlight) spacecraft in the side thrust configuration is unsatisfactory for the Comet Halley rendezvous mission.

- A propulsion system specific weight of the order of 30 kg/kWe is required to perform an attractive Comet Halley Mission. The 10 VDC internal fuel propulsion system exceeds this value for all varients evaluated. Except for the HPD propulsion system in the ALS launch configuration, all other propulsion system varients evaluated have specific weights less than 30 kg/kWe. This advantage allows for probable growth in propulsion system weight.

- The lightest 10 VDC internal fuel propulsion system specific weight is 31.2 kg/kWe for the single loop U-233 fueled varient. This value will increase to about 33.2 kg/kWe (34.4 kg/kWe for a U-235 fueled system) in order to provide the high redundancy multiple radiator loop propulsion system required by mission planners.

- The requirement that a portion of the net spacecraft be located immediately behind the propellant tanks is undesirable. Although employed to achieve center of gravity control in this study, splitting the net spacecraft is unacceptable in current NASA/JPL programs.

C. A space shuttle launch imposes performance penalties of 1.7 kg/kWe to 6.4 kg/kWe for these spacecraft designs.

D. The use of U-233 fueled reactors offer performance advantages of 5 kg/kWe, over U-235 fueled reactors, except for the 10 VDC internal fuel reactor spacecraft, where the performance advantage is only 1.0 kg/kWe.

F. The use of AC pumps minimizes pump power cable weights and EM pump power conditioning weights.

G. Spacecraft lengths above 20 meters cause launch vehicle integration problems.

6.0 RECOMMENDATIONS

A. The net spacecraft requires improved definition.

- Definition of the science payload, for comet rendezvous and planetary exploration missions in the 1980 to 1990 time period, will assist in resolution of the weight and volume of the science components that must be located at the end of the spacecraft.

- Definition of data handling and communications will define antenna size and power requirements. Antenna size limits imposed by spacecraft diameter or radiation shielding constraints will dictate both communication and data storage/ handling requirements.

- The propulsion system startup/restart subsystem is currently assumed to be a part of the net spacecraft. This subsystem definition is required, including weight and volume estimates of major components.

- Large amounts of electric power are potentially available at the end of the electric propulsion phase of the mission. The potential for the science payload to use this power, together with the impact of its use on spacecraft design, such as increased shielding weight, should be assessed.

B. This study provided a preliminary assessment of the impact of the Space Shuttle launch on the spacecraft design and arrangement. Additional work is recommended in the following areas:

- A more precise definition of the maximum size spacecraft (power level) that is compatible with shuttle payload volume and weight limits is required. This spacecraft must also meet the mission specific weight performance requirements. The scope of this effort should include both high L/D cylindrical, and conical-cylindrical spacecraft configurations. Foldable spacecraft and/or deployable power conditioning radiators should be assessed. This effort could result in the deliniation of a nominal

spacecraft compatible with both shuttle launch and spiral (low thrust) earth escape.

- Direct injection to earth escape would require the use of the shuttle with a kick stage of the Centaur class. Studies are required to define a NEP spacecraft and a chemical kick stage with the same shuttle launch. The feasibility of earth orbital assembly of a NEP spacecraft and a kick stage should be assessed.

C. The main, high voltage power conditioning requires improved definition in terms of weight and efficiency. This should be accomplished by building and testing prototypical units, since further analytical studies are of relatively low value at this time. The design and fabrication of these conditioners should use the power conditioning technology and components developed under the Solar Electric Propulsion System Technology (SEPST) at JPL, and should finalize the maximum allowable power conditioning temperature.

D. Powerplant part load operation requires definition in the areas of:

- The powerplant control system.
- The mission coast phase and post rendezvous phase, where the ion engines are not in operation; since these mission phases can encompass 5,000 to 10,000 hours, where the reactor is operated at 30 percent to 50 percent of full thermal power, the impact of the part-load-operation on reactor lifetime should be emphasized.
- Powerplant operation near earth; the heat rejection system is sized for deep space operation. The impact of near earth operation with higher effective sink temperatures should be assessed, particularly for low thrust spiral escape missions.

Potential effects include decreased electronics reliability if full power operation is employed, and increased mission time if reduced power levels are employed near earth in order to maintain electronic component temperatures within design limits.

E. The design and operation of the propellant feed system should be investigated, particularly for the side thrust spacecraft. It is expected that the degree to which uniform propellant flow can be maintained from the two tanks will have a significant effect on the TVC system requirements, particularly its axial translation. It is also desirable to investigate the location of the redundant ion engines within the thruster array.

F. Further studies in the use of the propellant as gamma shielding are recommended. Specifically, these studies should investigate:

- The impact of lower integrated dose limits
- The impact of the high dose rates present at the end of the mission on the performance of the science payload, even though the integrated dose limit is met by the shield design.
- Propellant tank design to minimize propellant ullage, tank weight, and to assure uniform propellant expulsion during the mission thrust phases and eliminate void formation in the propellant during the mission coast phase.

G. The definition of the 10 VDC Flashlight Reactor spacecraft assumed separate ion engine power condition in order to minimize the number of PC units, and therefore, PC and spacecraft weight. This approach requires the use of a separate fuse for each TFE, to provide protection for the electrical system in the event of a TFE short-to-ground. This technique requires further evaluation in terms of gradual TFE failure (short) as opposed to a sudden, total short-to-ground, and its effect on both the Flashlight Reactor and the spacecraft electrical system.

7.0 NEW TECHNOLOGY

No new technology items have been identified.

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